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STOL TACTICAL AIRCRAFT INVESTIGATION-EXTERNALLY BLOWN FLAP. VOLUME I. CONFIGURATION DEFINITION

Herschel G. Owens, et al

Rockwell International Corporation

Prepared for:

Air Force Flight Dynamics Laboratory

April 1973

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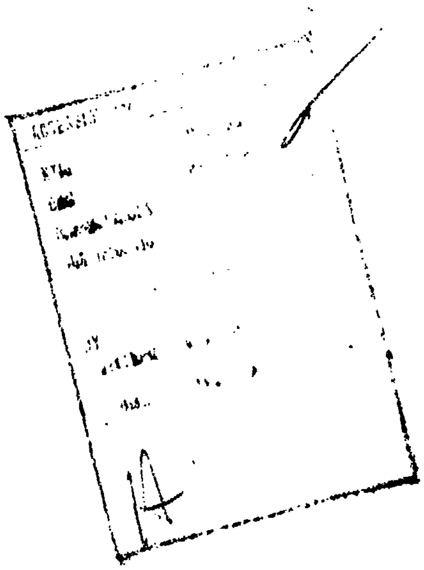

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13. ABSTRACT

The basic objective of the work reported herein was to provide a broader technology base to support the development of a medium STOL Transport (MST) airplane. This work was limited to the application of the externally blown flap (EBF) powered lift concept.

The technology of EBF STOL aircraft has been investigated through analytical studies, wind tunnel testing, flight simulator testing, and design trade studies. The results obtained include development of methods for the estimation of the aerodynamic characteristics of an EBF configuration, STOL performance estimation methods, safety margins for takeoff and landing, wind tunnel investigation of the effects of varying EBF system geometry parameters, configuration definition to meet MST requirements, trade data on performance and configuration requirement variations, flight control system mechanization trade data, handling qualities characteristics; piloting procedures, and effects of applying an air cushion landing system to the MST.

From an overall assessment of study results, it is concluded that the EBF concept provides a practical means of obtaining STOL performance for an MST with relatively low risk.

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STOL TACTICAL AIRCRAFT INVESTIGATION–
EXTERNALLY BLOWN FLAP

Volume I

Configuration Definition

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APRIL 1973

FOREWORD

This report was prepared for the Prototype Division of the Air Force Flight Dynamics Laboratory by the Los Angeles Aircraft Division, Rockwell International. The work was performed as part of the STOL tactical aircraft investigation program under USAF contract F33615-71-C-1760, project 643A0020. Daniel E. Fraga, AFFDL/PTA, was the Air Force program manager, and Garland S. Oates, Jr., AFFDL/PTA, was the Air Force technical manager. Marshall H. Roe was the program manager for Rockwell.

This investigation was conducted during the period from 10 June 1971 through 9 December 1972. This final report is published in six volumes and was originally published as Rockwell report NA-72-868. This report was submitted for approval on 9 December 1972.

This technical report has been reviewed and is approved.

E. J. Cross

E. J. Cross, Jr.
Lt Col, USAF
Chief, Prototype Division

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The technology of EBF STOL aircraft has been investigated through analytical studies, wind tunnel testing, flight simulator testing, and design trade studies. The results obtained include development of methods for the estimation of the aerodynamic characteristics of an EBF configuration, STOL performance estimation methods, safety margins for takeoff and landing, wind tunnel investigation of the effects of varying EBF system geometry parameters, configuration definition to meet MST requirements, trade data on performance and configuration requirement variations, flight control system mechanization trade data, handling qualities characteristics, piloting procedures, and effects of applying an air cushion landing system to the MST.

From an overall assessment of study results, it is concluded that the EBF concept provides a practical means of obtaining STOL performance for an MST with relatively low risk. Some improvement in EBF performance could be achieved with further development - primarily wind tunnel testing. Further work should be done on optimization of flight controls, definition of flying qualities requirements, and development of piloting procedures. Considerable work must be done in the area of structural design criteria relative to the effects of engine exhaust impingement on the wing and flap structure.

This report is arranged in six volumes:

Volume I - Configuration Definition

Volume II - Design Compendium

Volume III - Performance Methods and Takeoff and Landing Rules

Volume IV - Analysis of Wind Tunnel Data

Volume V - Flight Control Technology

Part I - Control System Mechanization Trade Studies
Part II - Simulation Studies/Flight Control System Validation
Part III - Stability and Control Derivative Accuracy Requirements and Effects of Augmentation System Design

Volume VI - Air Cushion Landing System Trade Study

This volume contains a summary of the part I baseline configuration and the refined configuration as optimized for minimum weight and the revised study requirements.

An appendix to this volume is issued separately, titled "Aerodynamic Trades of Flap and Roll Control System". This appendix is presented to provide the aerodynamic data needed to make the design choice between double and triple slotted flaps and between a roll control system with BLC or without BLC for the baseline configuration definition in Volume I.

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LIST OF SYMBOLS

ac	alternating current
ACLS	air cushion landing system
ADF	automatic direction finder
ADS	aerial delivery system
AEO	all engines operating
AGE	aerospace ground equipment
ALT	altitude, feet
AM	amplitude modulated
APU	auxiliary power unit
AR	aspect ratio
BCA	best cruise altitude, feet
BFDW	basic flight design weight, pounds
BLC	boundary layer control
BLM	best loiter mach number
BP	buttock plane
BPR	bypass ratio
BTC	bus tie contactor
C_{af}	chord of aft flap, inches
C_{ff}	chord of forward flap, inches
CA	converter assembly
CBR	California bearing ratio
CEF	critical engine failed
CEP	circular error probability
cfm	cubic feet per minute
CG	center of gravity
CLASS	close air support system
cp	center of pressure
CSD	constant speed drive
CTOL	conventional takeoff and landing
dc	direct current

DLC	direct lift control
EBF	externally blown flap
ECS	environmental control system
F	Fahrenheit
F_{T_u}	ultimate tensile stress, pounds/square inch
FAC	forward air controller
FEBA	forward edge of battle area
FM	frequency modulated
$F_{N_{max}}$	maximum thrust, pounds
FPR	fan pressure ratio
fps	feet per second
FS	fuselage station
g	gravity
GAG	ground-air-ground
gal	gallons
GE	General Electric
gpm	gallon per minute
h	altitude, feet
HF	high frequency
hr	hour
IBF	internally flown flaps
IDG	integrated drive generator
IFF	identification friend or foe
IGE	in ground effect
IP	identification point
IR	infrared
K_T	stress concentration factor
KEAS	knots equivalent airspeed
kva	kilovolt ampere
LDW	landing design weight, pounds
LN_2	liquid nitrogen

M	mach number
$M_{x,y,z}$	moment about X, Y, Z axis
MAC	mean aerodynamic chord, inches
MLDW	maximum landing design weight, pounds
MNFW	minimum flying weight, pounds
MODW	maximum over load design weight, pounds
mph	miles per hour
MST	medium STOL transport
N	load factor
Ng	gust load factor
n mi	nautical mile
N_Z	vertical load factor
OGE	out of ground effect
OLOGS	open loop oxygen - generating system
OP	outer panel
OPR	overall pressure ratio
OWE	operating weight empty
P/L	payload, pounds
P&W	Pratt & Whitney
PFCS	primary flight control system
PNdb	perceived noise level
psf	pound per square foot
psi	pound per square inch
q	dynamic pressure, pounds/square feet
R/S	rate of sink, feet/second
RPA	rational probability analysis
rpm	revolution per minute
SAM	surface-to-air missile
SCAS	stability and control augmentation system
SFCS	secondary flight control system

SLS	sea level static
STOL	short takeoff and landing
SUP	supervisory panel
T/W	thrust-to-weight ratio
TAC	tactical air command
TAI	tactical aircraft investigation
TIT	turbine inlet temperature
T.O.	takeoff
TOGW	takeoff gross weight, pounds
TRU	transformer rectifier units
UHF	ultra-high frequency
V_G	design equivalent gust velocity, feet/second
V_H	level flight maximum speed, feet/second
V_L	limit speed, feet/second
V_{LO}	liftoff velocity, feet/second
V_S	stall velocity, feet/second
VHF	very high frequency
VOR	variable omnirange
VSCF	variable-speed constant frequency
VSPEP	vehicle sizing and performance program
VT	vectored thrust
W_a	engine airflow, pounds/second
W/S	wing loading, pounds/square feet
WL	water line
WRP	wing reference plane
WUTO	warmup and takeoff
X_{cp}	center of pressure location, inches
X/C	percent chord
X_V	displacement along vehicle X-axis, inches

Y_F	displacement along vehicle Y-axis, inches
Z_F	displacement along vehicle Z-axis, inches

b	wing span, feet
C	chord force, pounds
c	local chord, inches
$\bar{c}$	mean aerodynamic chord, inches
C_D	nondimensional total drag, $\frac{D}{qS}$
C_L	nondimensional total lift, $\frac{L}{qS}$
$C_{L_{max}}$	nondimensional maximum lift, $\frac{L_{max}}{qS}$
C_L	nondimensional total lift L/qS
C_l	nondimensional rolling moment $\mathcal{L}$/qSb
C_{l_p}	nondimensional roll damping derivative $\frac{d\mathcal{L}}{dp} \cdot \frac{1}{qSb}$
C_m	nondimensional pitching moment $\frac{M}{qS\bar{c}}$
C_N	nondimensional normal force $\frac{N}{qS}$
C_n	nondimensional yawing moment $\mathcal{N}$/qSb
C_{n_r}	yawing damping derivative $\frac{dn}{dr} \cdot \frac{1}{qSb}$
C_μ	blowing coefficient, $\frac{T}{1/2\ V^2S}$
D	total drag, pounds
F	resultant force, pounds
f	function of
g	acceleration of gravity, feet/second^2

h	altitude above terrain, feet
$\dot{h}$	altitude rate, feet/second
I	moment of inertia, pounds-foot-second^2
L	total lift, pounds
$\mathcal{L}$	rolling moment, foot-pound
ℓ	tail arm, distance from $0.25c_W$ to $0.25c_{H \text{ or } V}$, feet
M	pitching moment, foot-pound
N	normal force, pound
$\mathcal{N}$	yawing moment, foot-pound
$\dot{p}$	angular roll acceleration, radians/second^2
p	angular roll rate, radians/second
$\dot{q}$	angular pitch acceleration, radians/second^2
q	angular pitch rate or freestream dynamic pressure, $1/2\rho V^2$, pounds/foot^2
r	angular yaw rate, radians/second
S	wing reference area, square feet
T	nozzle exhaust thrust, pounds
T_{PE}	nozzle exhaust thrust per engine, pounds
U	X body axis velocity, feet/second
V	freestream velocity - sometimes with subscript t for total, feet/second
W	aircraft weight, pounds
w	velocity along the Z body axis, feet/second
$\dot{w}$	acceleration along the positive Z body axis, feet/second^2
Y	side force, pounds
y_f	spanwise location of outboard edge of flap, inches
y_j	spanwise location of jet engine center line, inches
Z_f	distance which flap trailing edge extends into jet plume perpendicular to engine center line, inches
z_H	height of horizontal tail above the wing, inches

α	angle of attack, degrees
α_{max}	angle of attack at maximum lift, degrees
β	angle of sideslip, positive nose left, degrees
Δ	increment due to power
δ	angular deflection of an undefined control surface, degrees
δ_a	differential aileron and/or roll spoiler deflection angle, degrees
δ_e	elevator deflection angle, degrees
δ_F	flap deflection angle, degrees
δ_H	horizontal stabilizer deflection angle, degrees
δ_N	deflection angle of engine exhaust deflector, degrees
δ_R	rudder deflection angle, degrees
δ_T	thrust nozzle vector angle, degrees
δ_{sp}	direct lift control (DLC) spoiler deflection angle, degrees
ϵ	downwash angle, degrees
ϕ	aircraft bank angle, degrees
θ	aircraft pitch attitude, degrees
Λ	wing sweep of 0.25 chord, degrees
ρ	density of freestream flow, slugs/foot^3
ω	frequency, radians/second
ζ	damping ratio

SUBSCRIPTS

$(\ )_o$	zero angle of attack
$(\ )_{PE}$	per engine
$(\ )_{TE}$	trailing edge
$\dot{\alpha}$	angle of attack rate

SECTION I

INTRODUCTION

The objective of the STOL tactical aircraft investigation (TAI) is the expansion of the technical base to support the development of a medium STOL transport (MST) with minimum risk. This technical base provides a means to generate design criteria, performance data, stability and control data, and prediction methods for application to and evaluation of a short takeoff and landing (STOL) tactical transport aircraft. This study is part of an Air Force program originating from the Prototype Division of the Air Force Flight Dynamics Laboratory (FDL) in which several contractors have participated, each working on the various propulsion-lift systems. The purpose of the present study is to develop the capability of evaluating proposed configurations of all reasonable concepts of STOL-type aircraft. This work has been focused primarily on those propulsion lift systems using the mechanical flap system with vectored thrust (VT), the externally blown flap (EBF), and the internally blown flap (IBF). The Rockwell International contract requires the development of data for the EBF propulsion-lift system only.

The general approach used by North American Rockwell to achieve the program objectives provides a logical sequencing of related tasks within a two-part program that would produce substantiated data and methods consistent with the Air Force overall program. The interrelationship of the various study phases is depicted in the task flow diagram, shown in figure 1, with the refined baseline configuration development.

This study was conducted in two parts with six tasks in each part. Part 1 (studies and analyses) identified missing aerodynamic performance and stability and control data to be acquired in the wind tunnel test program of part 2, (small-scale model tests). During part 1, a specific task to develop a baseline configuration was developed for evaluating STOL performance ground rules, establishing optimum propulsion system requirement, and developing trade data for comparing air cushion landing system (ACLS) with conventional landing gear designs. The baseline configuration development in part 1 continued into part 2 of the study with a refined configuration definition using data developed from other program tasks. This refined configuration is optimized for minimum weight and meets the study mission requirements and STOL performance criteria as revised by the AFFDL/PTA Project Office. This volume of the STOL tactical aircraft investigation final report contains a summary of the results of the part 1 baseline configuration definition and the refined configuration definition from part 2.

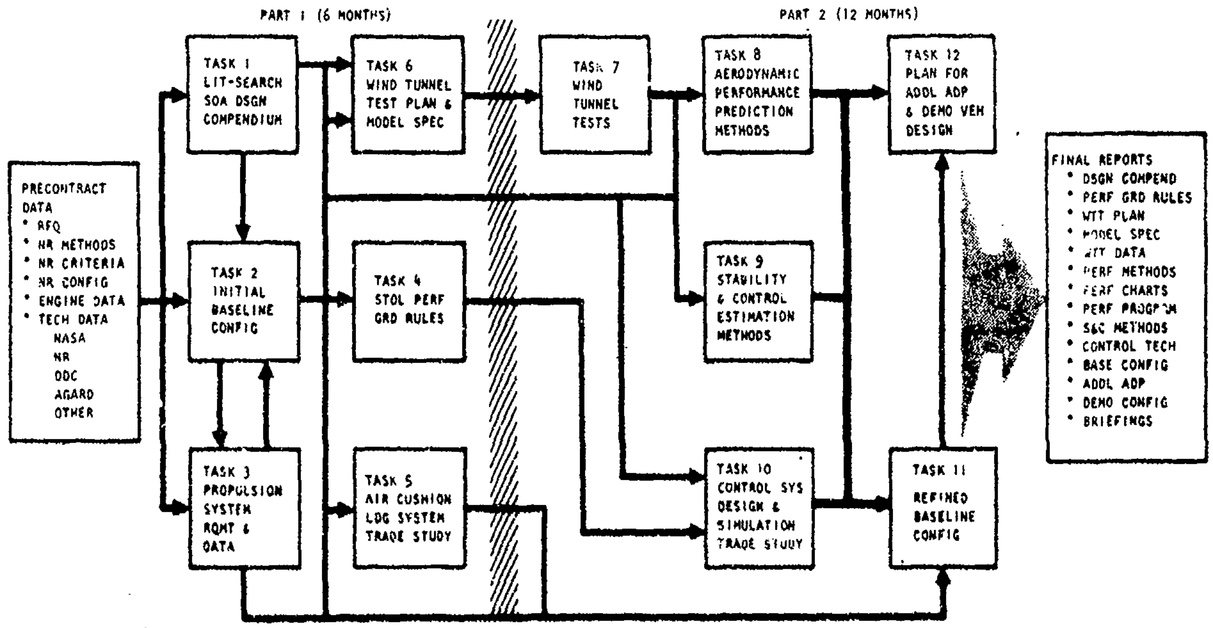


Figure 1. Task Flow Diagram

SECTION II

PRELIMINARY BASELINE CONFIGURATION SUMMARY

A summary of the baseline configuration study performed in part 1 of the STOL TAI program is presented in this section. This task provided for the development of a baseline configuration designed to meet the MST requirements as outlined in appendix I of the statement of work. The configuration developed under this task serves as the baseline vehicle for the various structure analysis studies and other pertinent trades study required by the STOL TAI contract.

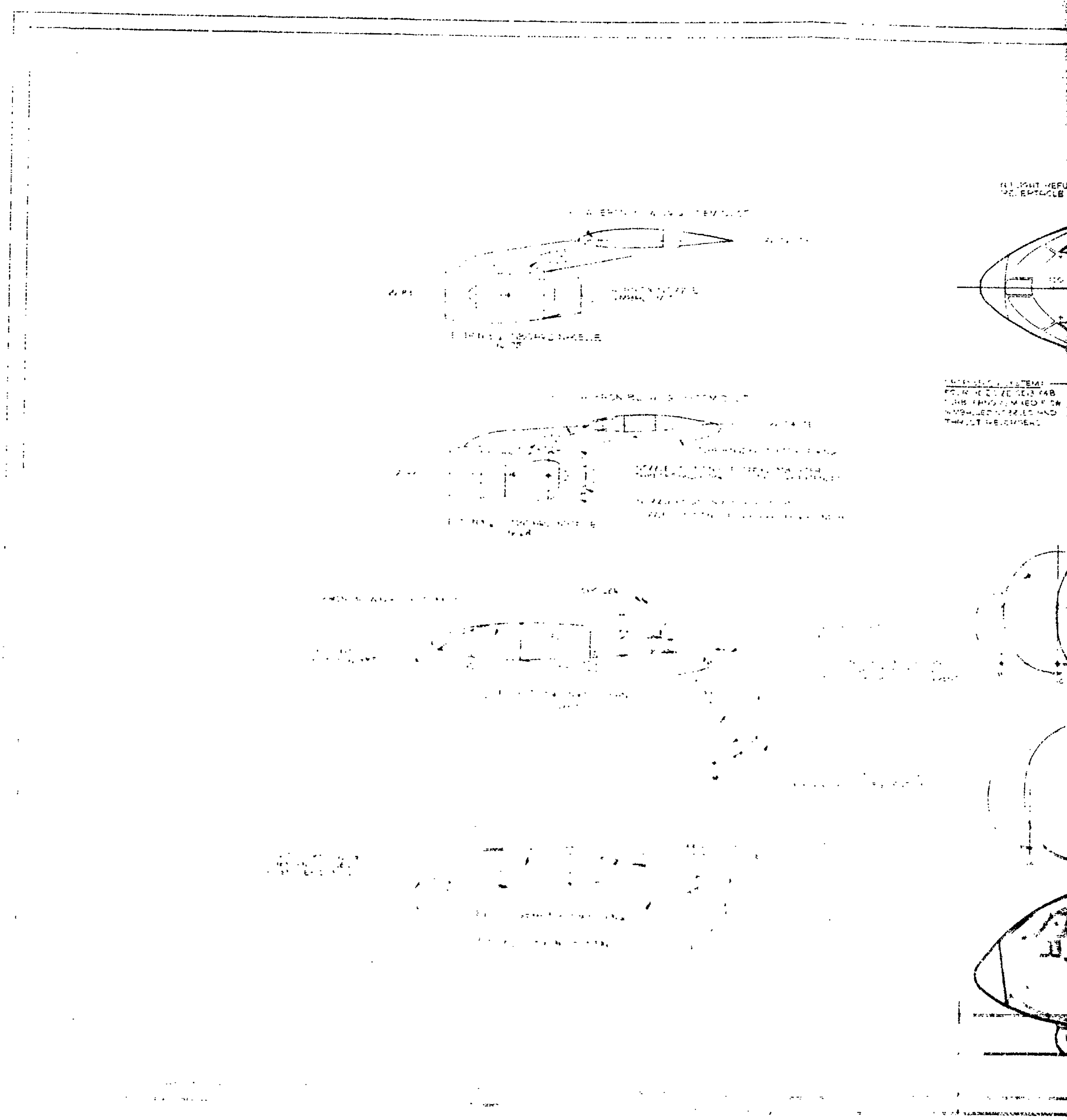
2.1 CONFIGURATION DEFINITION

The task II baseline configuration is shown in figure 2. This vehicle has a takeoff gross weight (TOGW) of 150,000 pounds, an initial wing loading of 89.28 pounds per square foot, are initial thrust-to-weight ratio of 0.54 at sea-level static (SLS), standard day conditions and carries a 28,000-pound payload on a 500-nautical-mile-radius mission at a 0.75M cruise speed. It is designed to take off and land over a 50-foot obstacle in 2,000 feet at a 2,500-foot altitude, 93.41° F hot day at midmission weight. The midmission wing loading is 82 pounds per square foot, and thrust-to-weight ratio is 0.53 at the specified altitude condition.

The fuselage has been sized based on the required 12 x 12 x 45-foot cargo bay. A 210-inch-diameter circular cross section is utilized because of pressurization requirements, with some flattening underneath to provide adequate ground clearance while maintaining the cargo floor height at 50 inches above the ground. The crew compartment is sufficient for a four-man crew, rest facilities, and avionics equipment. The aft fuselage has been faired as sharply as possible and yet avoid separation drag.

The wing selected has an aspect ratio of 8 and sweep of 25 degrees at the quarter chord. The airfoil section is a 63A200 series with 10 percent thickness at the tip and 14 percent thickness at the root. The trailing edge has been extended from X_F = 240.00 inboard to reduce the effective thickness at the fuselage centerline to 12.0 percent. The selected wing planform and thickness are the result of several wing design iterations to improve weight at constant performance.

The wing incorporates full-span spoilers, powered leading edge slats, and double-slotted flaps. The chord of the inboard two-thirds of the slat is 20 percent of the wing chord, and this section can be extended to 30 degrees below the wing chord plane. The chord of the outboard portion of the slat is 30 percent of the wing chord, and this portion can be extended to 45 degrees below the wing chord plane.



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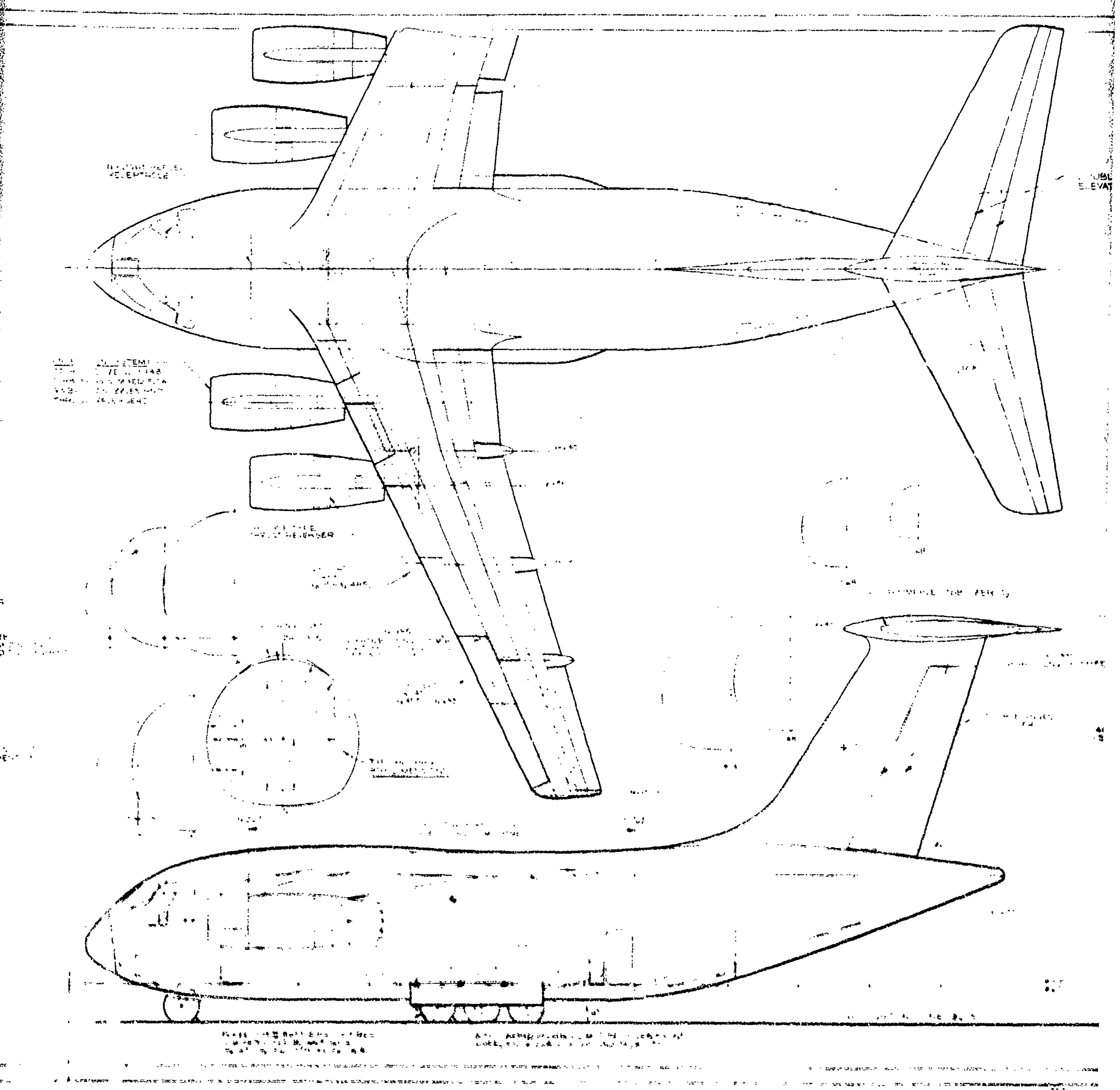

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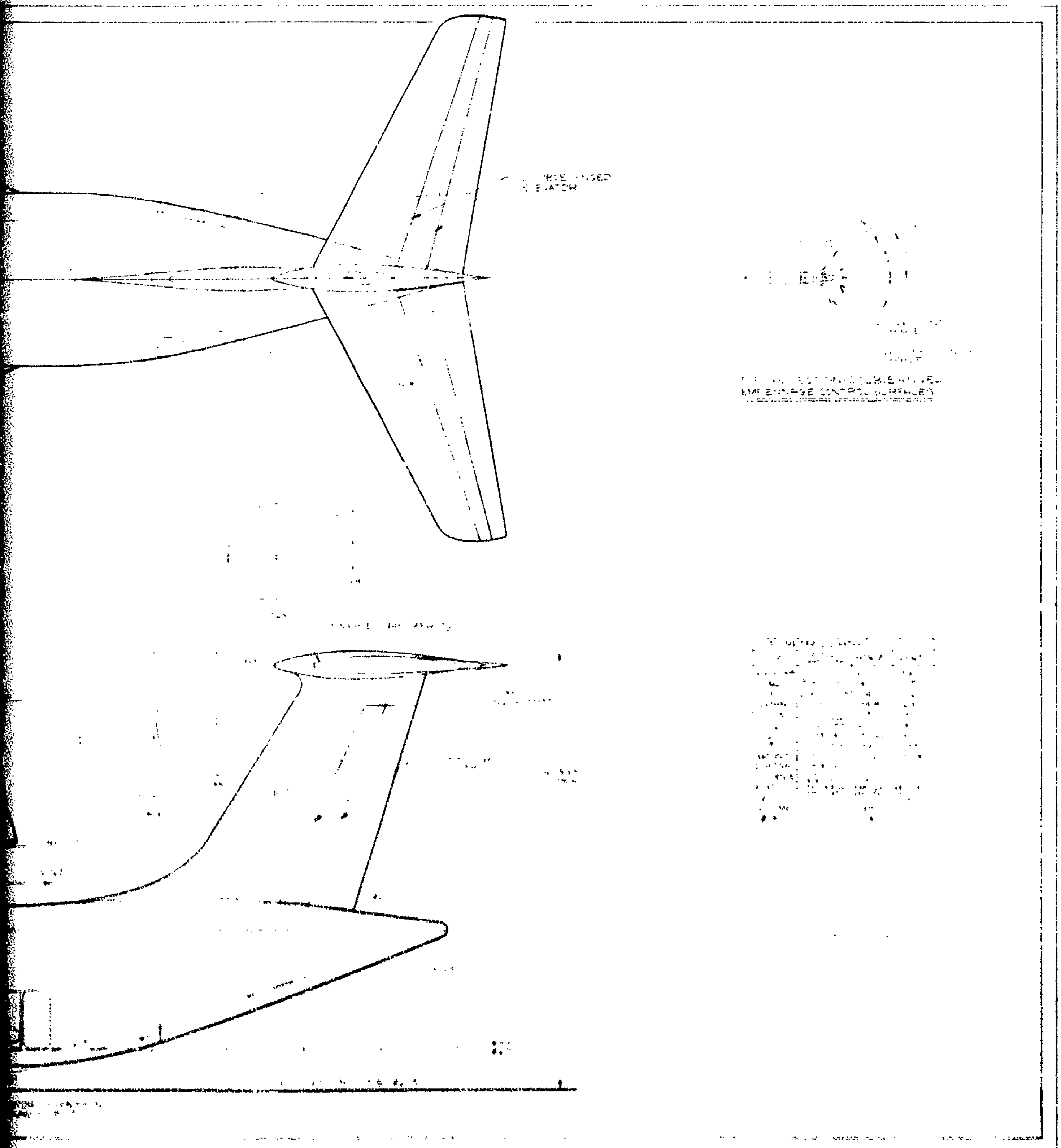


Figure 2. Preliminary Basepoint Configuration

The double-slotted flaps are rather large, 36 percent of the wing chord, with the forward flap 17 percent and the aft flap 19 percent. The forward flap maximum extension is 45 degrees, and the aft flap maximum extension is 60 degrees. The flap system is divided into three spanwise segments. The aft segment of the inboard flap can be articulated ±25 degrees from a fully extended position for direct drag control during final landing approach.

The aileron is a double-panel flaperon system with the total system chord equal to 36 percent of the wing chord. The aft panel is 19 percent of the wing chord and acts as a conventional plain-hinge aileron during cruise flight and climb or descent with flaps retracted. When the flaps are extended, the forward flaperon panel is extended 25 degrees to fixed position, and the aft panel is deflected to a neutral position of 60 degrees. From this position, it can be deflected ±25 degrees for roll control, in combination with spoilers. Aileron blowing is provided, using primary engine fan air, to provide increased aileron effectiveness at high deflection angles. The aileron blowing system is sized to a total C_μ of 0.03, available with three or four engines, with a nozzle system that is normally half open to divide the aileron blowing equally between left and right ailerons. The nozzles are controllable to full open or full closed to provide 100 percent of the available blowing to the aileron that is deflected downward.

The maximum fuel capacity of the integral wing fuel tanks is 56,000 pounds, with the rear spar at the 61 percent chord and a modified wing center-section box.

The engine selected is a 90 percent size GE13/F4B turbofan with a 7.8 bypass ratio. For the baseline configuration, the inboard engine has been located one diameter from the fuselage, and the two engines are separated by 1-1/2 diameters. The nozzle exit is placed on the same station as the wing leading edge at the engine centerline. A gimbaled nozzle deflects both the hot and cold exhaust air 15 degrees up or down. During the takeoff ground roll, the exhaust is deflected 15 degrees down then changed to 15 degrees up at liftoff and during climb until the flaps are retracted. The exhaust is also deflected up 15 degrees during landing approach when the flaps are extended. With flaps up during climb, cruise, or descent, the exhaust is undeflected. Cascade-type thrust reversers are provided for the fan air only.

The main landing gear is a three-wheel tandem design retracted into pods off the lower shoulder of the fuselage in a manner conventional for high-wing transport aircraft. The 50 x 22, type III tires have been sized for a CBR 6 and 200 passes at a TOGW of 150,000 pounds. Design sink speed is 15 feet per second at midmission weight. Reduced sink speed is used at higher air vehicle weight. The nose gear is a two-wheel dual, forward retracting design. Although analysis shows that a smaller tire could be used on the nose gear, type III tires have been used identical to the main gear tires to simplify logistics and servicing. The aft wheel in the three-wheel tandem main gear can

be swiveled so that, during taxi, the airplane will turn as if the main gear were a two-wheel tandem design. The front wheel on each main gear is powered, using the APU, for ground mobility without using primary flight engines.

The empennage is a T-tail arrangement with the horizontal surface placed slightly higher and further forward than conventional T-tails because of the strong downwash characteristics of the EBF system. Both the vertical and horizontal tail have double-hinged control surfaces with hinges at 55 percent and 75 percent chord. Each segment deflects 25 degrees independently, making a total of 50 degrees relative to the centerline for the aft segment when both segments are fully deflected. Both segments are used for control during STOL operations; however, for conventional flight, the forward segment is locked and the aft segment only is used for control. The forward horizontal surface can also be adjusted +10 degrees and -20 degrees for airplane trim.

2.2 DESIGN REQUIREMENTS

Requirements specified in the text and in the statement of work have been summarized in table I for the basepoint and trade-offs in terms of mission, performance, and design characteristics.

2.3 MISSION DESCRIPTIONS

The 500-nautical-mile radius mission and the 2,600 nautical-mile deployment range mission are basepoint design requirements, and are illustrated in figures 3 and 4. Study requirements permit the wing fuel tank capacity to be limited to that required for the 2,600 nautical-mile range mission, and additional range is achieved by use of fuselage tanks, external tanks, or flight refueling. The study has shown that (1) TOGW is determined by the 500-nautical-mile radius mission requirement, (2) wing area is determined by fuel required for the 2,600-nautical-mile deployment mission, and (3) lift-off velocity for STOL performance is determined by one-engine-out climb gradient requirement of 3 degrees.

2.4 TAKEOFF AND LANDING CRITERIA

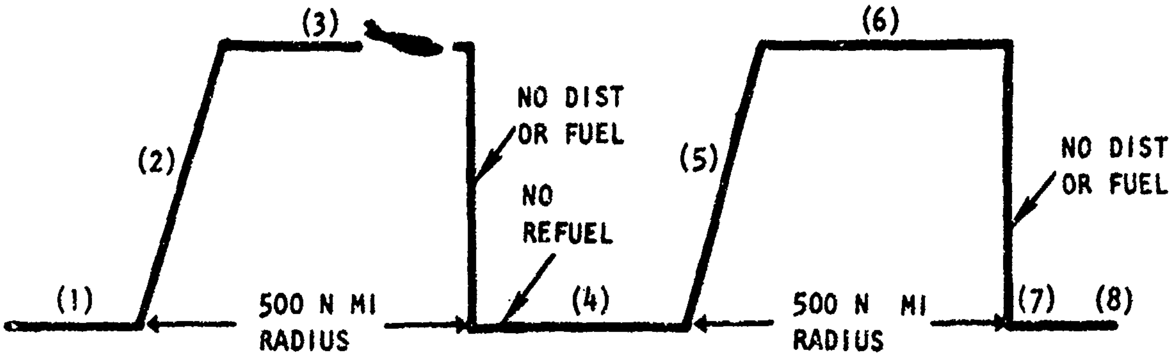
The STOL takeoff and landing criteria used for the preliminary baseline configuration are depicted in figure 5. The takeoff and landing maneuvers are performed at a 2,500-foot 93.4° F day condition. The predominant sizing conditions from these criteria are the 3-degree climb gradient with one engine out, landing gear down, out of ground effect, and a takeoff distance of 2,000 feet over a 50-foot obstacle. These conditions determine the thrust levels required, thus vehicle engine size. The landing maneuver did not influence vehicle size.

TABLE I. MST DESIGN REQUIREMENTS

Parameter	Basepoint	Trade-offs
Missions		
Design radius	500 n mi	750 n mi
Payload (both ways)	28,000 lb	
Maneuvering limit	3g	
Unrefueled range (2.5g): 2,600 n mi with reduced cargo and 3,600 n mi with no cargo		
Performance		
Cruise speed	0.75M at $h \geq 20{,}000$ ft	0.85M
Penetration speed at SL for 50 n mi		400 knots
(This requirement refers to a terrain-following environment with associated avionics. This is not baseline, only a trade-off.)		
T.O. distance over 50 ft 2,500 ft alt/93° F	2,000 ft	±500 ft
Design Characteristics		
Max design TOGW	160,000 lb	
Bay size	12 x 12 x 45 ft	12 x 12 x 55 ft
Crew - pilot, copilot, navigator, loadmaster		
Propulsion - derivative engines using existing core permitted		
Miscellaneous features:		
Aerial refueling		
Pressurized crew and cargo compartments (scale C-130 equipment)		
Wheel drive for ground handling at remote site		
Landing gear for CBR 6 allowing for 200 passes and lower CBR at reduced payload; design sink speed of 15 fps		
Antiskid brakes		
Minimum AGE utilization and no special facilities		
Vulnerability/survivability; maximum protection and fail-safety		
Satisfactory engine-out control		
Maximum evasive maneuverability		
Paratroop and equipment airdrop capability. Equipment airdrop up to 400 knots		
463L compatibility		
Self-contained off-load capability		
Drive-On loading capability (not drive-thru)		
24-hour continuous operations		
No noisier than C-130E, i.e., 115 Pndb at 500 ft sideline		
Three-engine takeoff capability from assault strip with sufficient fuel for 500 n mi return leg, without cargo		
As a general rule, fixed equipment shall be C-130E equipment or scaled C-130E equipment		

Special Comment

TAC is interested in airlifting 58,000 pounds (self-propelled howitzer) on a one-time basis (2.5g). This would, in all probability, be from a conventional concrete runway with whatever radius and landing performance that can be obtained. A similar interest is in airlifting 44,000 pounds (2.5g) on an infrequent basis (but not one-time). These situations shall be examined for the baseline design.



LEG	OPERATION	PWR	TIME (MIN)	SPEED	ALT	DIST
(1)	START & TAKEOFF	NRP & MAX	5	-	SL	0
(2)	CLIMB	MAX CLIMB	TBD*	-	TO BCA***	TBD
(3)	CRUISE	AS REQD	TBD	0.75M	BCA	TBD
(4)	START & TAKEOFF	NRP & MAX	5	-	SL	0
(5)	CLIMB	MAX CLIMB	TBD	-	TO BCA	TBD
(6)	CRUISE	AS REQD	TBD	0.75	BCA	TBD
(7)	LOITER	AS REQD	30	BLM**	SL	0
(8)	RESERVES (5% INITIAL FUEL)	-	-	-	-	-

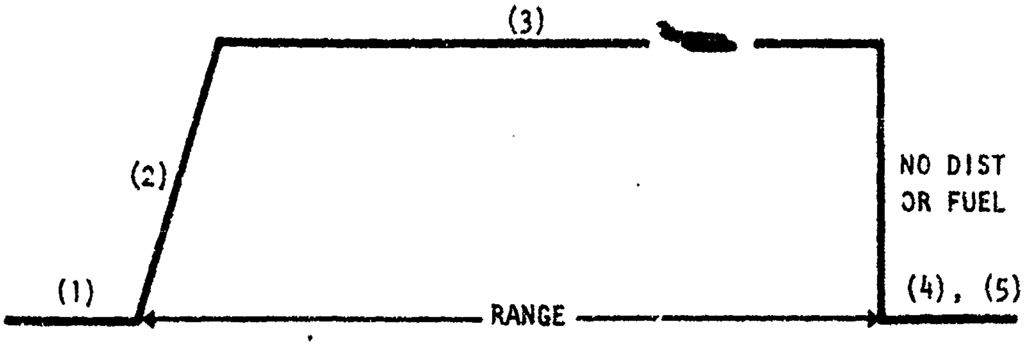
PAYLOAD: DESIGN PAYLOAD OUTBOUND & RETURN (28,000 POUNDS)

STANDARD ATMOSPHERE

SERVICE TOLERANCE: 5% ON ALL FUEL FLOWS

* TBD = TO BE DETERMINED
** BLM = BEST LOITER MACH NUMBER
*** BCA = BEST CRUISE ALTITUDE

Figure 3. Design Mission Profile



LEG	OPERATION	PWR	TIME (MIN)	SPEED	ALT	DIST
(1)	START & TAKEOFF	NRP & MAX	5	-	SL	0
(2)	CLIMB	MAX CLIMB	TBD	-	TO BCA ***	RANGE****
(3)	CRUISE	AS REQD	TBD	BCM*	BCA	
(4)	LOITER	AS REQD	30	BLM**	SL	0
(5)	RESERVES (5% INITIAL FUEL)	-	-	-	-	-

STANDARD ATMOSPHERE

SERVICE TOLERANCE: 5% ON ALL FUEL FLOWS

* BCM = BEST CRUISE MACH NUMBER
** BLM = BEST LOITER MACH NUMBER
*** BCA = BEST CRUISE ALTITUDE
**** RANGE = 2600 N MI WITH REDUCED PAYLOAD (2.5 G)
= 3600 N MI WITH ZERO PAYLOAD (2.5 G)

Figure 4. Deployment Mission Profile

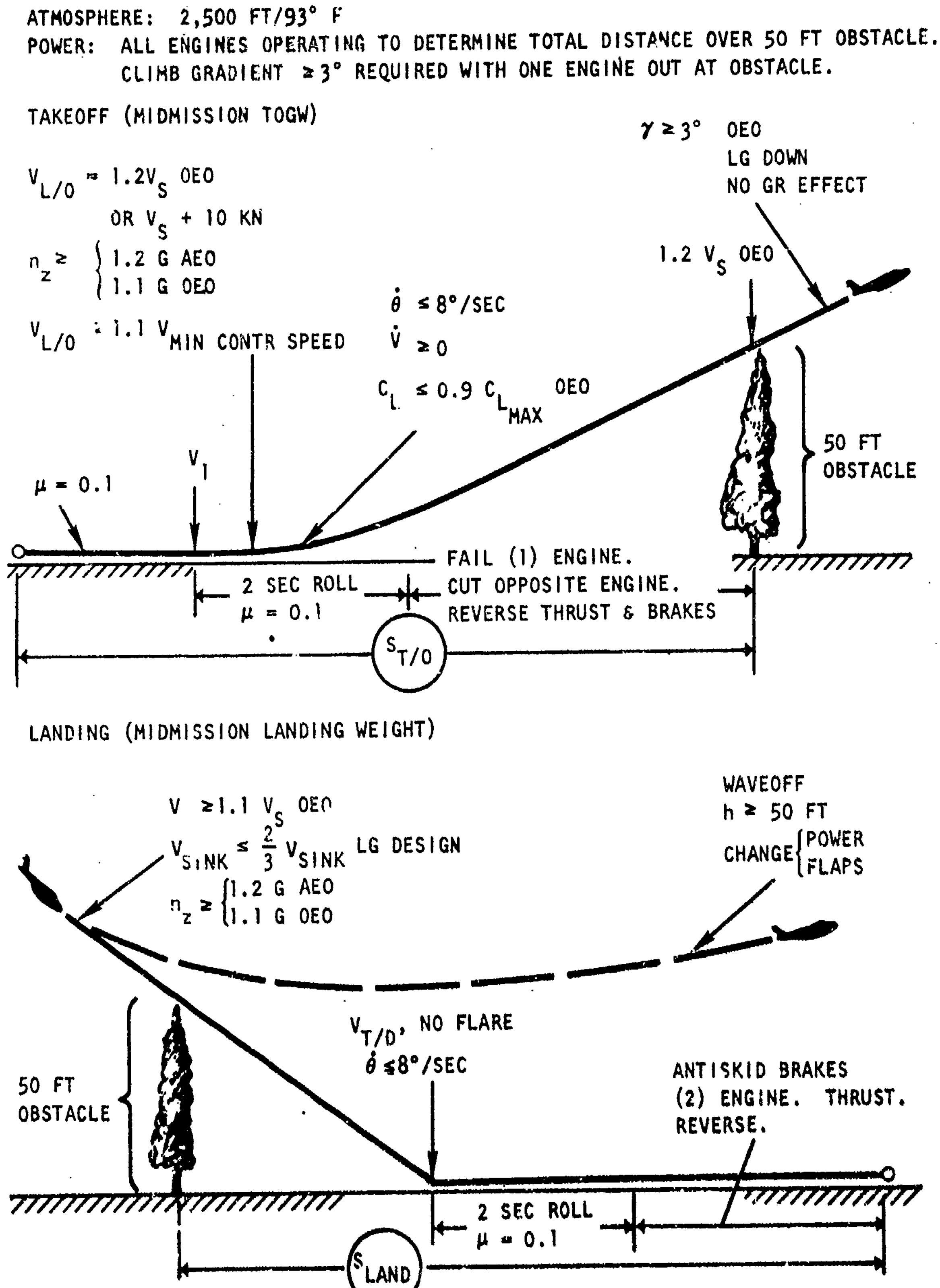


Figure 5. Takeoff and Landing Criteria

2.5 WEIGHT SUMMARY

The weight data presented in tables II and III reflect the weight and balance, respectively, of the D-516-2 basepoint configuration. The TOGW and payload are given at 150,000 pounds and 28,000 pounds, respectively, with fuel being the variable. It can be seen that 26,535 pounds of usable fuel are available. The avionics equipment list used in the weight analysis is presented in table IV.

The criteria used to determine the weight and balance of the basepoint configuration were based on the following design assumptions.

2.5.1 DESIGN ASSUMPTIONS

The flutter and maximum design q of 383 pounds per square foot are based on a cruise speed of 0.75M at 20,000 feet and constant q to sea level where $V_H = V_L$ or $M_H = 0.51$. The maximum tension allowable of 0.85 F_{T_U} was assumed, in lieu of fatigue allowable data. Only the inboard portion of the wing torque box was affected by this constraint.

2.5.1.1 Surface Control System

The flight control system weight is based on an assumed fully powered, irreversible system including a stability augmentation system. The primary flight control system (PFCS), which includes pitch, roll and yaw control surfaces, is fairly complicated due to assumed low- and high-speed requirements. These requirements are as follows:

Surface	Deflection	Rate, Max
Elevator - double-hinged	25/50 deg	50 deg/sec
Rudder - double-hinged	25/50 deg	50 deg/sec
Spoiler	60 deg	120 deg/sec
Direct drag control	±25 deg	50 deg/sec

The secondary flight control (SFCS) consists of double-slotted trailing edge flaps and powered leading edge slats. The manual flight control system includes the pilot's input controls, cables, push-pull rods, bellcranks, and supports. The automatic flight control system includes a pilot-assist function, stability augmentation, and a fail-operational system.

The electronic equipments (controllers, sensors, and transducers) are based on data derived from in-house bomber studies.

TABLE II. WEIGHT SUMMARY

	Pounds
Total structure	66,995
Wing group	23,430
Tail group - horizontal	3,870
- vertical	3,500
Body group	22,210
Alighting gear group - main	5,642
- auxiliary	998
Surface controls	2,505
Engine section or nacelle group	4,840
Propulsion group	(16,835)
Engine (as installed) GE13/F4B 90%	13,620
Accessory gearboxes and drives	
Air induction system	
Exhaust system	1,175
Cooling and drain provisions	
Lubricating system	
Fuel system	1,140
Engine controls	125
Starting system	325
Propeller installation	
APU	450
Fixed equipment	(10,135)
Instruments	900
Hydraulic and pneumatic group	990
Electrical group	1,310
Electronics group	1,060
Armament provisions	
Furnishings	4,045
Air conditioning equipment	1,410
Photographic	
Auxiliary gear	100
Ground mobility	320
Total weight empty	93,965
Crew	860
Fuel	
Internal	26,535
Trapped	350
Oil	200
Engine	
Trapped	
Armament	
LN_2	90
Payload	28,000
Equipment	
Total Useful Load	56,035
Takeoff gross weight	**150,000**
Flight design gross weight	
Landing design gross weight	

TABLE III. BALANCE

Item	Weight (lb)	Arm	Moment
Structures group			
Wing	23,430	469.0	10,988,670
Tail - horizontal	3,870	1,099.0	4,253,130
- vertical	3,500	1,244.0	4,354,000
Body	22,210	594.2	13,197,182
Alighting - main	5,642	544.3	3,070,941
- auxiliary	998	127.1	126,846
Surface controls	2,505	686.2	1,718,931
Engine section	4,840	313.0	1,514,920
Total	66,995	(585.5)	39,224,620
Propulsion			
Engines	13,620	293.8	4,001,280
Aileron BLC	1,175	400.5	470,555
Fuel system	1,140	387.3	431,500
Engine control	125	230.0	28,750
Starting system	325	307.4	99,890
APU	450	460.0	207,000
Total	16,835	(311.8)	5,248,975
Equipment			
Instrument	900	237.8	213,986
Hydraulic	990	469.3	464,605
Electrical	1,310	336.3	440,501
Electronics	1,060	190.5	201,946
Furnishing	4,045	336.7	1,362,148
Air conditioning	1,410	359.1	506,368
Aux gear	100	514.8	51,475
Ground mobility	320	465.2	148,850
Total	10,135	(334.5)	3,389,879
Weight empty, gear down	93,965	(509.4)	47,863,474
Weight empty, gear up	93,965	(509.4)	47,869,385
Useful load			
Crew	860	123.0	105,780
Fuel - Usable	26,535	440.0	11,675,400
- Trapped	350	440.0	154,000
Oil	200	291.0	58,200
LN_2	90	335.0	30,150
Payload	28,000	471.3	13,196,000
Total	56,035	(450.1)	25,219,530
Takeoff gross weight, gear down	150,000	(487.2)	73,083,004
Takeoff gross weight, gear up	150,000	(487.3)	73,088,915

TABLE IV. AVIONICS EQUIPMENT LIST

Function	Identification	Weight (lb)	Volume (cu ft)	Power (watts)
HF comm	ARC-125	38	0.95	500*
VHF AM comm	ARC-115	8	.105	50
VHF FM comm	ARC-131	39	.48	100
UHF comm	ARC-144	34	.775	210
Public address	AIC-13 (65PK)	28.5	.96	53
Intercomm	C-6533/ARC (4)	7.4	.192	15
Intercomm station	C-6624/AIC-25 (2)	5.4	.094	13
Microphone/headset	H-101/A or equiv	-	-	-
Voice enco	KY-28 KY-57	18		33
Crash recorder	(B-1 or equiv)	18	.48	105
Tacan	ARN-91	36	.60	163
Glide slope marker beacon	R-844/ARN-88	11		9
UHF/ADF	ARA-50	17	.592	55
VOR/LOC rec	ARN-82	12		28
Station keeping	ARN-169	57		300
Radar beacon	ARN-154	5		35
IFF	APX-90	17	.233	100
Secure encoder	KIT-1A/TSEC	15	.29	30
Air data computer	Arinc 575	11	.213	35
Flight director computer	CPU-80	6	.08	25
Inertial navigation	LN-30	12.5	.166	150
Attitude/heading reference	Sperry GSS	23	.514	98
Doppler radar	APN-185	77		350
Multimode radar	R-87	221	14.66	1,000
Radar altimeter	APN-198	7.5	.092	55
Terminal landing aid system		40	.42	75
Integrated weight and balance system		27	.142	30
Gust alleviation		24	.31	20
	Total	815.3	22.348	3,637

* ARC-1.23 power = receiver 150 watts, transmitter = 740 watts
Redundancy should be considered

2.5.1.2 Propulsion Group

The basic engine weight was obtained from a General Electric engine brochure. Thrust reverser weight was estimated at 10 percent of the basic engine weight. The vector nozzle weight increment was estimated with the following equations:

$$3.16\ (W_a) \text{ or thrust/weight} = 250$$

Both produce approximately the same weight. These relations were obtained from prior nozzle studies.

2.5.1.2.1 Starting System

The starting system consists of engine-mounted air turbines with cross ducting between the engines. The air turbines can be powered by either the air vehicle APU or an external air source.

2.5.1.2.2 Auxiliary Power Unit (APU)

The APU installation was assumed to be the same as that proposed for the CX-6. The APU is an AiResearch CTCP-85 series unit.

2.5.1.2.3 Fuel System

The fuel is contained in integral wing tanks. Sufficient return-home fuel is assumed to be contained in the wing center section which is considered to be protected from ground fire by fuselage structure below it. A liquid nitrogen inerting system is also provided.

2.5.1.2.4 Boundary Layer Control (BLC)

The BLC system weights were estimated using both the air vehicle configuration layout and a schematic of the system. The ducts were assumed to be constructed of 0.040-inch aluminum. An installation factor of two was applied to the bare duct weight to account for expansion joints, fittings, elbows, supports, etc. This installation factor was derived from prior ducting studies.

2.5.1.3 Fixed Equipment

2.5.1.3.1 Instrument Group

The weights allocated for this group includes the visual indication system for the following functions:

Flight path control
Engine operation
Fuel quantity
Cabin pressure
Surface control position
Compartment temperature
Miscellaneous

The weights include indicators, sensors, transmitters, amplifiers, and wiring. Previous transport studies and current USAF transports were used for instrument and installation data.

2.5.1.3.2 Hydraulic System

A 4,000 psi, four-system hydraulic fluid power system is provided. Each system is a completely independent system, and no physical transfer of fluid from one system to another is possible. Basically, four systems supply power to the primary flight control system. The utility function arrangement was not considered at this time. However, each system supplies both primary and utility functions. To preclude total fluid loss in case of line or component failure in the utility functions, isolation valves are included.

The distribution system weight calculation was based on flow and length from the power generation to the user.

2.5.1.3.3 Electrical System

The electrical system consists of two parallel systems with power source provided by four 20 kva IDG's. These generators are mounted on each accessory gear drive. Generator-CSD cooling is accomplished through an oil-to-fuel heat exchanger. The dc provisions include transformers, transformer/rectifiers, and the distribution system.

A 15kva hydraulic-driven generator is provided for the emergency system.

2.5.1.3.4 Furnishing Group

The furnishing group includes provisions for personnel accommodation, cargo handling, seat tracks, litter supports, and soundproof thermal

insulation. Oxygen provisions are for 6-hour duration. Food and drinks are based on 24-hour duration.

Item	Weight (lb)
Personnel Accommodation	
Seats - pilot, copilot, and navigator	165
Load master	27
Chemical toilet provision	40
Waste receptacle	5
Tracks and supports	152
Jump signal provision	15
Static line retriever	20
Food and drink provision	60
Food (1 lb/meal x 6 meal/man x 4)	24
Drink (1 gal/man x 8.3 lb/gal x 4)	33
Supports	3
O_2 Provisions	141
6 hr duration can be accomplished by 0.2-10 liter bottles	-40
O_2 bottles	50
Lines, valves, etc	10
Cargo compartment provisions	41
Total	625
Miscellaneous Furnishings	
Platform and ladder	30
Instrument board	90
Data cases	25
Consoles	100
Thermal insulation	200
Soundproofing	400
Partitions	25
Floor covering	25
Total	895

Item	Weight (lb)
Emergency Equipment	
Fire detection system (nacelles	100
Fire detection system (fuselage)	75
First aid Kit	10
Ditching station, life raft provisions	160
Portable fire extinguisher	30
Total	375

2.5.1.3.5 Passive Defense

Approximately 15 square feet per crewman of dual hardness armor plate is provided. The area density to provide protection for a .30-caliber ball at 100 yards range and 30 degrees obliquity is approximately 6 psf. These requirements were assumed noting the probability of fuselage structures, and subsystem equipment, etc, providing some degree of protection such that the range and degree of obliquity selected is reasonable.

Item	Weight (lb)
Crew protection (15 sq ft x 6.0 x 4)	360
Equipment (16 sq ft x 6.0)	96
Armor glass (lower window only)	50
Miscellaneous	44
Total	550

2.5.1.3.6 Cargo Handling

Based on 463L cargo loading system, the total cargo handling weight is as follows:

Item	Weight (lb)
Cargo tiedown	78
Side rails	135
Center rails	201
Rollers and conveyers	422
Aerial delivery	90
Winch and self-contained unloading system	273

Item	Weight (lb)
Paratroop provision	30
Gear storage	195
Pallet safety	27
Tiedown receptacles	149
Total	1,600

2.5.1.3.7 Electronic System

The total weight of the electronic system was based on the following:

Item	Weight (lb)
Equipment list	815.3
Antenna addition	50.0
Installation	194.7
Total	1,060.0

The equipment list (table IV) includes some flight control functions; e.g., director computer and air data computers. Automatic flight control equipments (e.g., controllers) are coded to surface controls.

2.5.1.3.8 Air Conditioning and Anti-Ice

The environmental system is assumed to be an air-cycled system rather than vapor-cycled as previously considered in the CX-6 studies. The weight of the system was based on an arithmetic mean of current applicable aircraft air conditioning system weights divided by approximate total airflows.

2.5.1.3.9 Ground Mobility

The ground mobility system consists of a hydraulic pump driven by the auxiliary power unit and a hydraulic drive unit at each of the main gear forward wheel.

The system is sized to the following conditions:

Dry paved runways
Paved slope, maximum of 3 degrees
2 g operating condition
Average forward velocity of 5 mph

		Weight (lb)
Weight Summary		
Power supply		(120)
Pump	57.0	
Pump disconnect	7.5	
Reservoir	4.3	
Filters	20.0	
Plumbing	22.2	
Miscellaneous (supports, dampers, etc)	9.0	
Drive unit (2)		(200)
Motor	34.0	
Mechanical drive	100.0	
Control(s)	32.0	
Shafting, connection, miscellaneous	34.0	
Total ground mobility		320

2.6 ENGINE DESCRIPTIONS

2.6.1 CANDIDATE ENGINES

A survey was conducted to identify engines in the thrust and availability category for application to the MST. A list of some of these engines and their basic characteristics are given in table V. Pratt and Whitney Aircraft (P&W), General Electric (GE), and Allison engines are shown in the categories of separate flow and mixed flow engines.

On the basis of cruise efficiency, engine thrust-to-weight ratio and engine diameter which influences nacelle size, weight, and drag, two P&W engines appeared to be likely candidates. On the same basis, four GE engines and three Allison engines were selected as candidates. A 90 percent GE 13/F4B engine was finally selected as the basepoint engine on the basis of availability, efficiency of cruise, size, and weight. Selection of the GE/F4B is not at this time considered to be the ultimately best choice, but is a reasonable choice for initial studies of a baseline configuration. Noise consideration, availability, cost, and overall airframe engine matching factors, when more thoroughly evaluated, could result in another engine being the best final selection.

The GE 13/F4B turbofan engine is representative of other high bypass ratio turbofans proposed by P&W and Allison that would provide the performance, low noise characteristics, and availability required by this program. The GE 13 series engines use the F101 basic gas generator (core) being developed for

TABLE V. MST CANDIDATE ENGINES

Mfr	Model	SLS ($F_{N_{max}}$)	SLS BPR	SLS FPR	SLS OPR	TIT_{max} (° F)	T/W	Max Dia (in.)	Core Engine
SEPARATE FLOW ENGINES									
P&W	STF337	20,029	5.47		19.0	2,400	7.42	61.3	
	STF342A	23,000	5.91		13.0		6.97	65.3	
	STF342B	23,000	5.91		15.0		7.12	65.4	
	STF342C	23,000	5.91		20.5		7.08	65.8	
	STF342D	23,000	3.00		15.0		6.95	56.5	
	STF342E	23,000	9.00		15.0		6.71	72.2	
	STF362	20,000	9.00				6.46	66.8	
	STF391	19,930	9.00				6.35	68.0	JTF22
	STF392	24,019	9.00				6.41	74.8	JTF22
GE	GE13/F3B	21,320	5.1	1.7	24.1		7.61	65.8	F101
	GE13/F2B	22,860	6.5	1.6	23.4		7.59	72.1	F101
	GE13/F4B	23,950	7.8	1.5	22.4		7.24	77.5	F101
Allison	PD351-2	21,650	6.25	1.65	24.8	2,465	8.52	60.6	
	PD351-3	21,650	4.6	1.65	24.8	2,465	8.63	59.0	
	PD351-4	21,650	4.0	1.7	24.8	2,465	8.73	57.2	
	PD351-5	21,650	8.0	1.45	21.8	2,465	8.33	66.0	
	PD351-48	21,650	5.25	1.65	24.8	2,465	8.22	62.0	Quiet Eng
	PD351-49	23,630	8.0	1.45	21.8	2,465	7.62	72.0	Quiet Eng
MIXED FLOW ENGINES									
P&W	STF402	19,751	5.0				7.60	60.0	JTF22
	STF403	19,381	3.8				6.86	60.0	JTF22
GE	GE13/F3A	21,000	5.0	1.6	23.8		7.50	65.8	F101
	GE13/F2A	22,640	6.4	1.5	23.3		7.52	72.1	F101

the B-1 air vehicle which incorporates the latest advanced technology. Since the major portion of the engine is already under development for a military program, a derivative engine would have an advantage relative to risk, availability, and cost.

Data on the 100 percent size GE 13/F4B engine were provided for preliminary analysis. This engine has a sea-level static thrust rating of 24,000 pounds and a bypass ratio of 7.8. The uninstalled dry weight is 3,310 pounds. Although the GE 13/F4B is designed as a separate flow turbofan, it was assumed that the mixed flow version for the EBF application could be made available without much change in performance.

2.7 PERFORMANCE

2.7.1 TAKEOFF PARAMETRIC STUDY RESULTS

The results of a parametric study conducted to establish sizing criteria are presented in figures 6 through 14. These figures show the effects of lift-off velocity and ground roll and air distance for a flap setting of $\delta f_1/\delta f_2$ of 20/40 degrees. In addition, data for three flap settings (10/20 degrees, 15/30 degrees, and 20/40 degrees) were calculated to examine flap angle effects. All takeoff evaluations are for 2,500-foot altitude, 93.41° F day (MIL-STD-210A hot atmosphere).

The principal ground rules considered for this study involved the establishment of the lift-off velocity which would satisfy the following criteria:

1. $V_{Lo} \geq 1.2$ Vs (one engine out) or V_S + 10K

2. $V_{Lo} \geq 1.1\ V_1$ for minimum control speed

3. V for $N_z \geq 1.2g$ all engines operating

   V for $N_z \geq 1.1g$ one engine out

4. V for climb gradient ≥ 3 degrees one engine out, gear down, and out of ground effect at 50-foot obstacle

5. V at 50-foot obstacle ≥ 1.2 V_S (one engine out)

6. Rolling coefficient μ_R = 0.1

7. $\dot{\theta} \leq 8$ degrees/second, $\theta = \alpha + \gamma$, α = angle of attack, γ = flight path angle

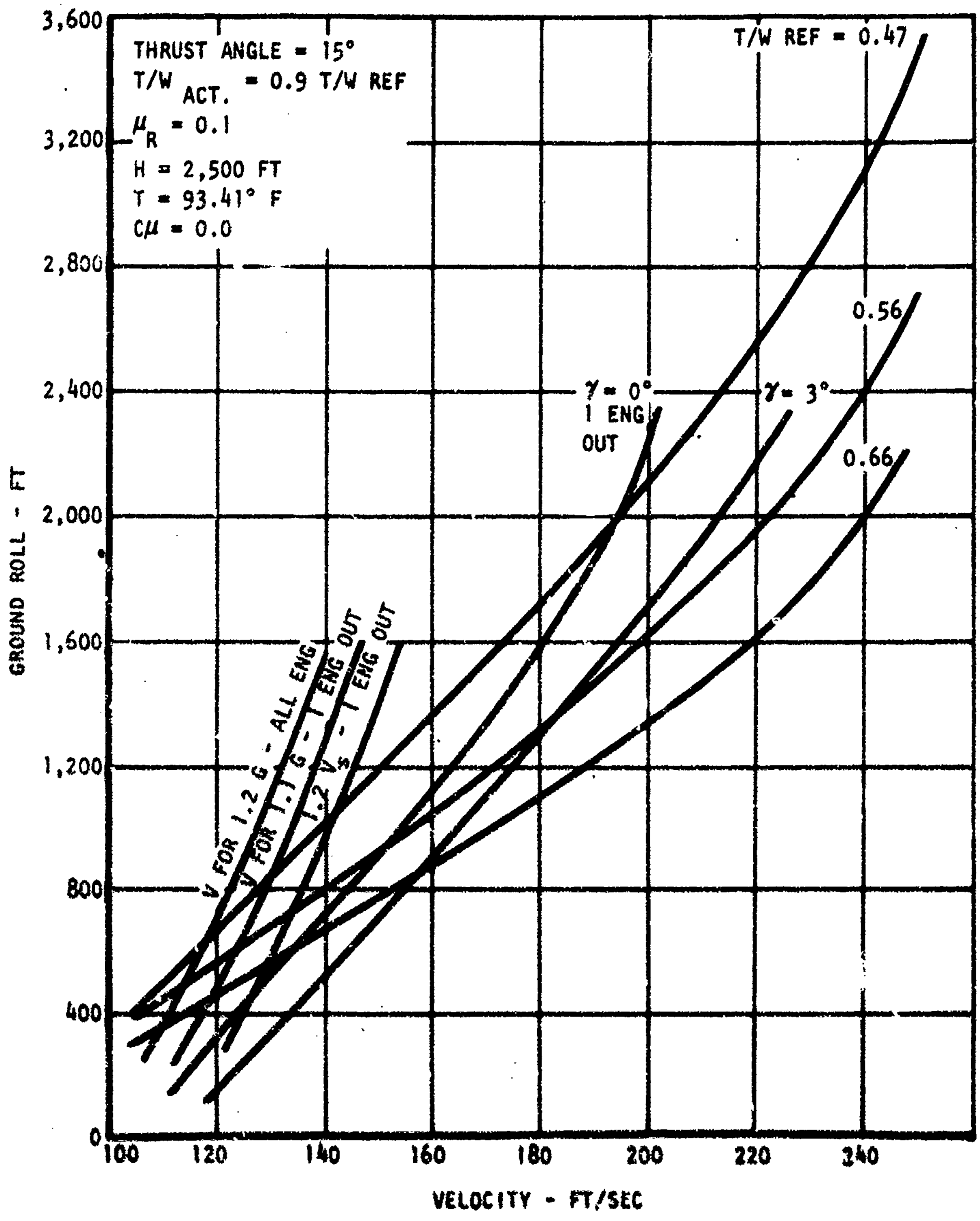


Figure 6. Ground Roll Versus Lift-Off Velocity - $\delta_{f_1}/\delta_{f_2}$ = 20°/40°, W_o/S = 70 lb/ft^2

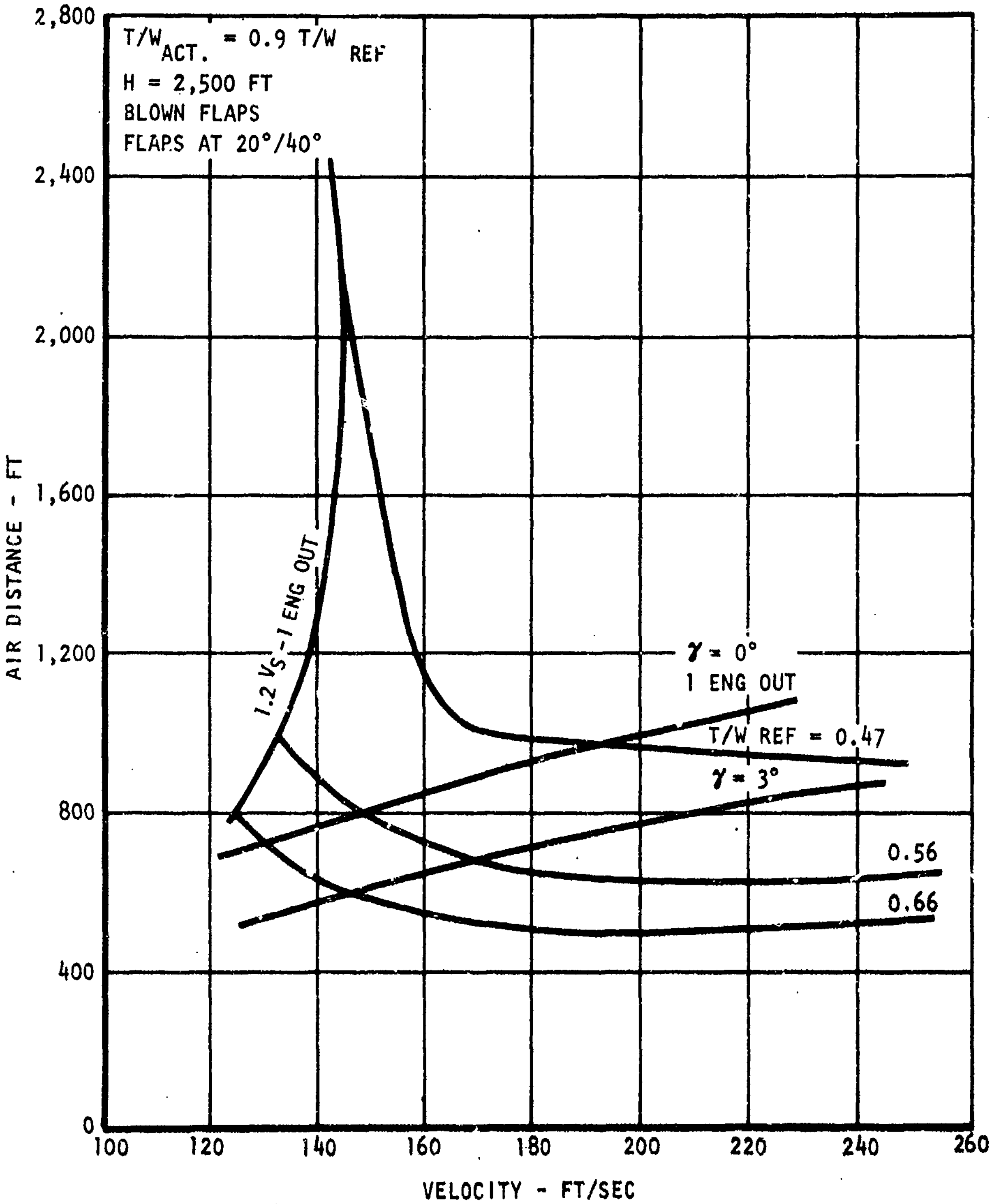


Figure 7. Air Distance Versus Lift-Off Velocity - $\delta_{f_1}/\delta_{f_2}$ = 20°/40°, W_o/S = 70 lb/ft^2

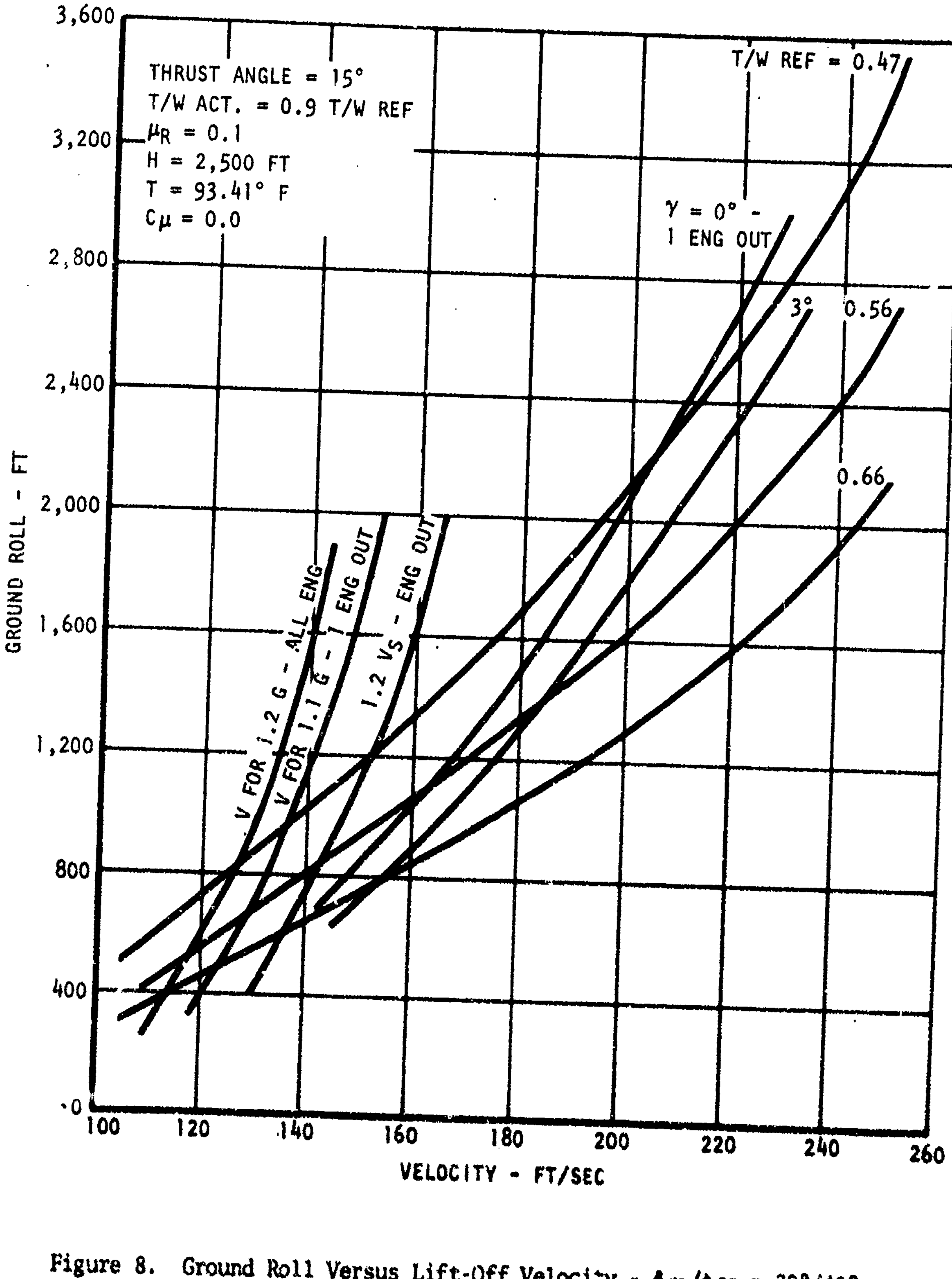


Figure 8. Ground Roll Versus Lift-Off Velocity - δ_{f1}/δ_{f2} = 20°/40°, W_0/S = 80 lb/ft^2

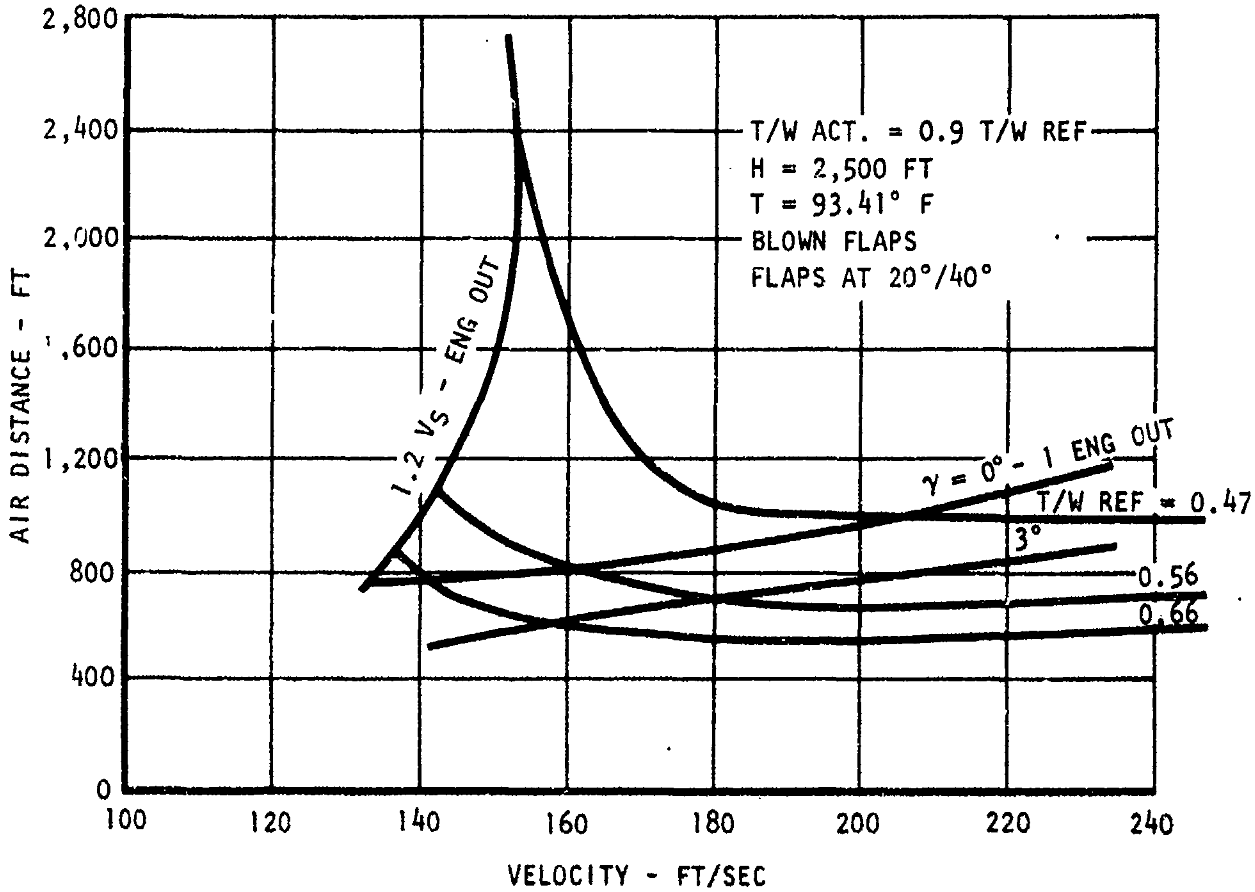


Figure 9. Air Distance Versus Lift-Off Velocity - δ_{f1}/δ_{f2} = 20°/40°, W_o/S = 80 lb/ft^2

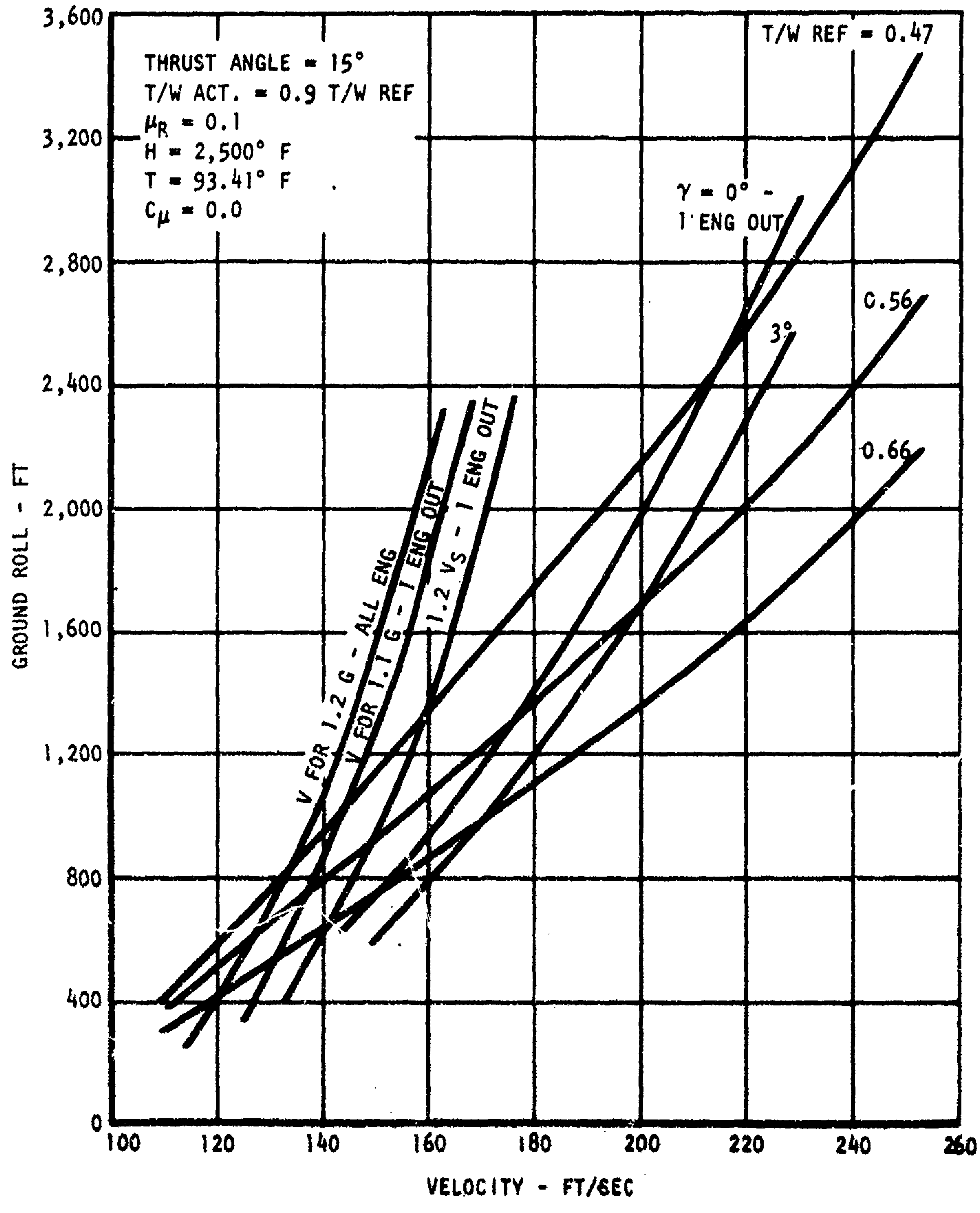


Figure 10. Ground Roll Versus Lift-Off Velocity - δ_{f1}/δ_{f2} = 20°/40°, W_O/S = 90 lb/ft^2

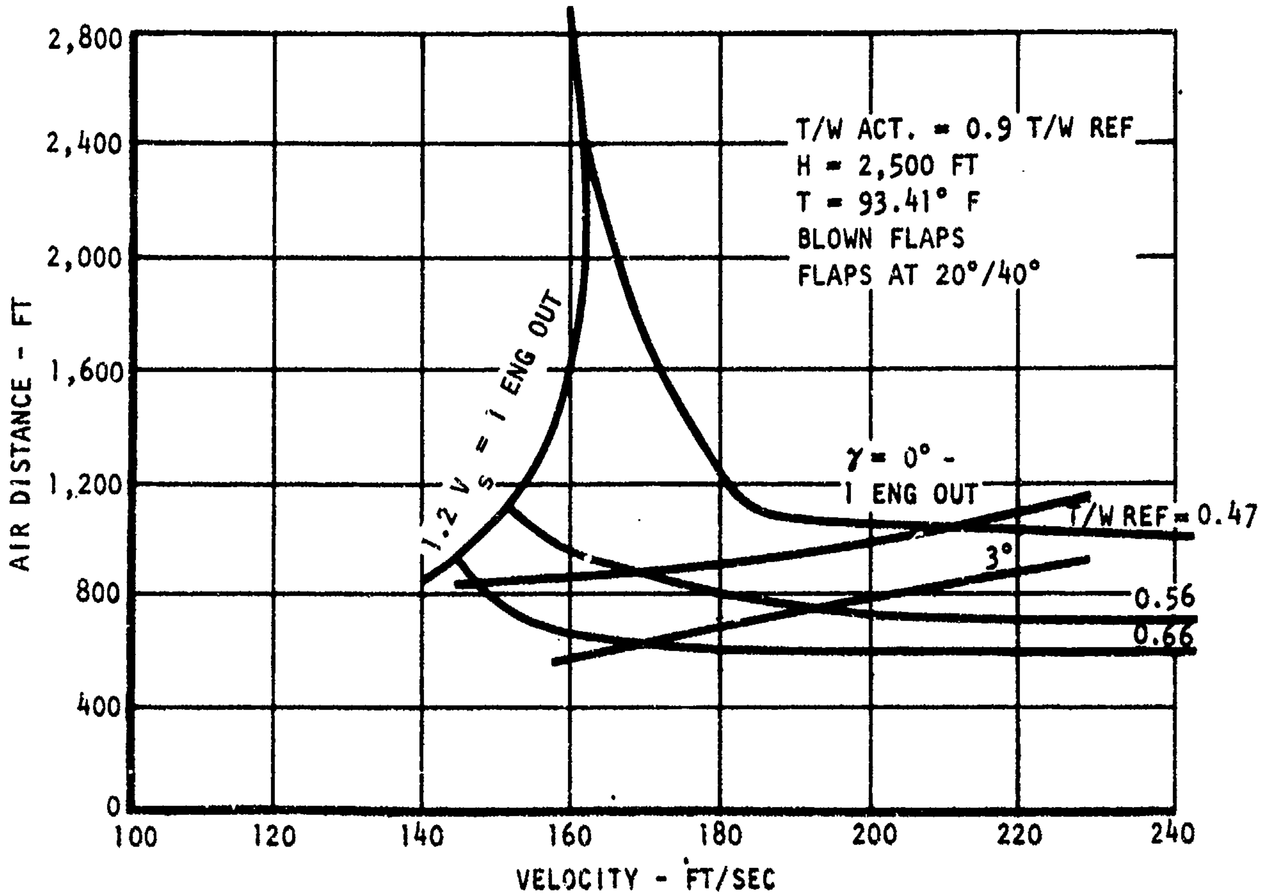


Figure 11. Air Distance Versus Lift-off Velocity - δ_{f1}/δ_{f2} = 20°/40°, W_o/S = 90 lb/ft^2

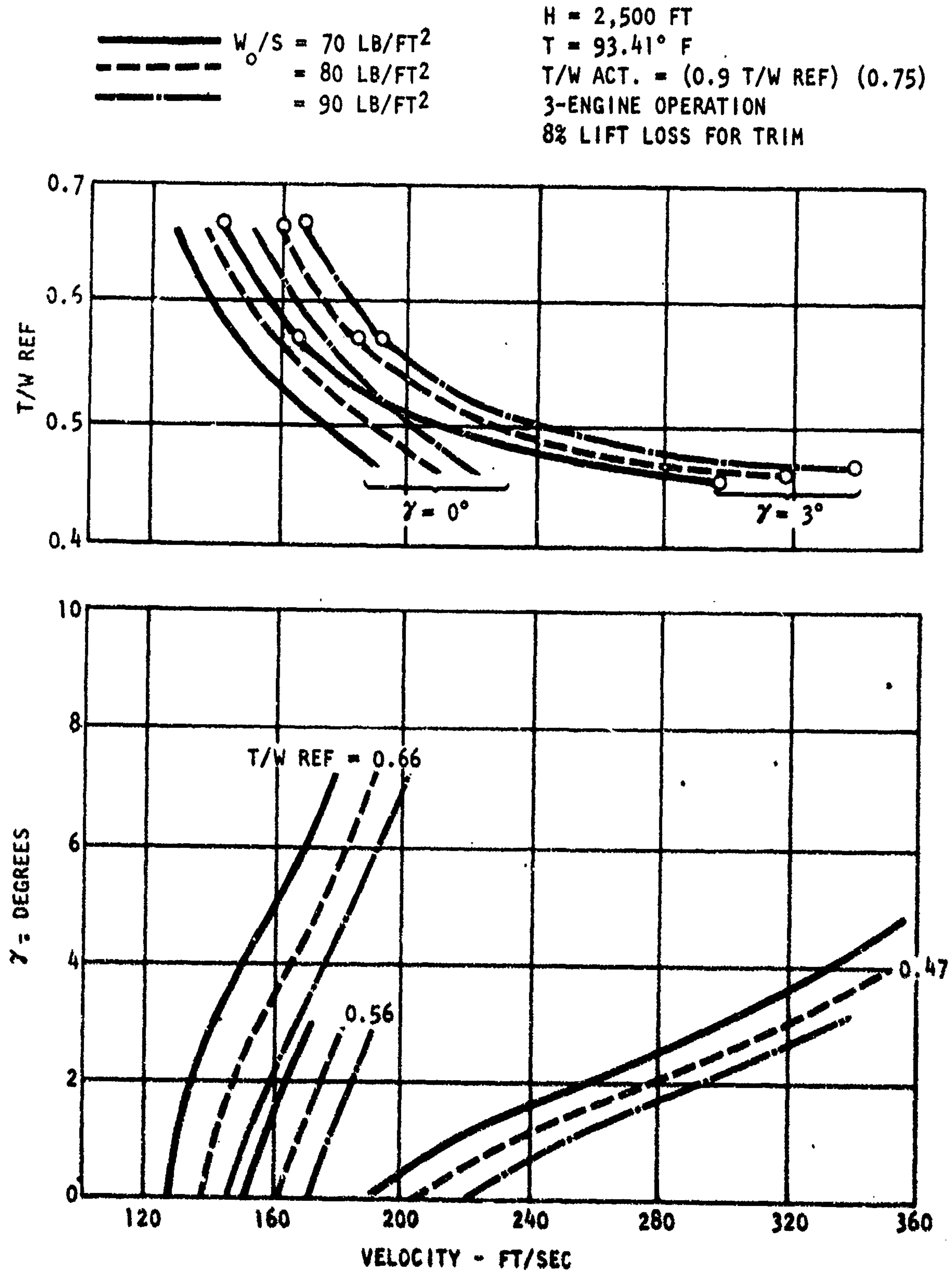


Figure 12. Obstacle Climb Gradient - One Engine Out - δ_{f1}/δ_{f2} = 20°/40°

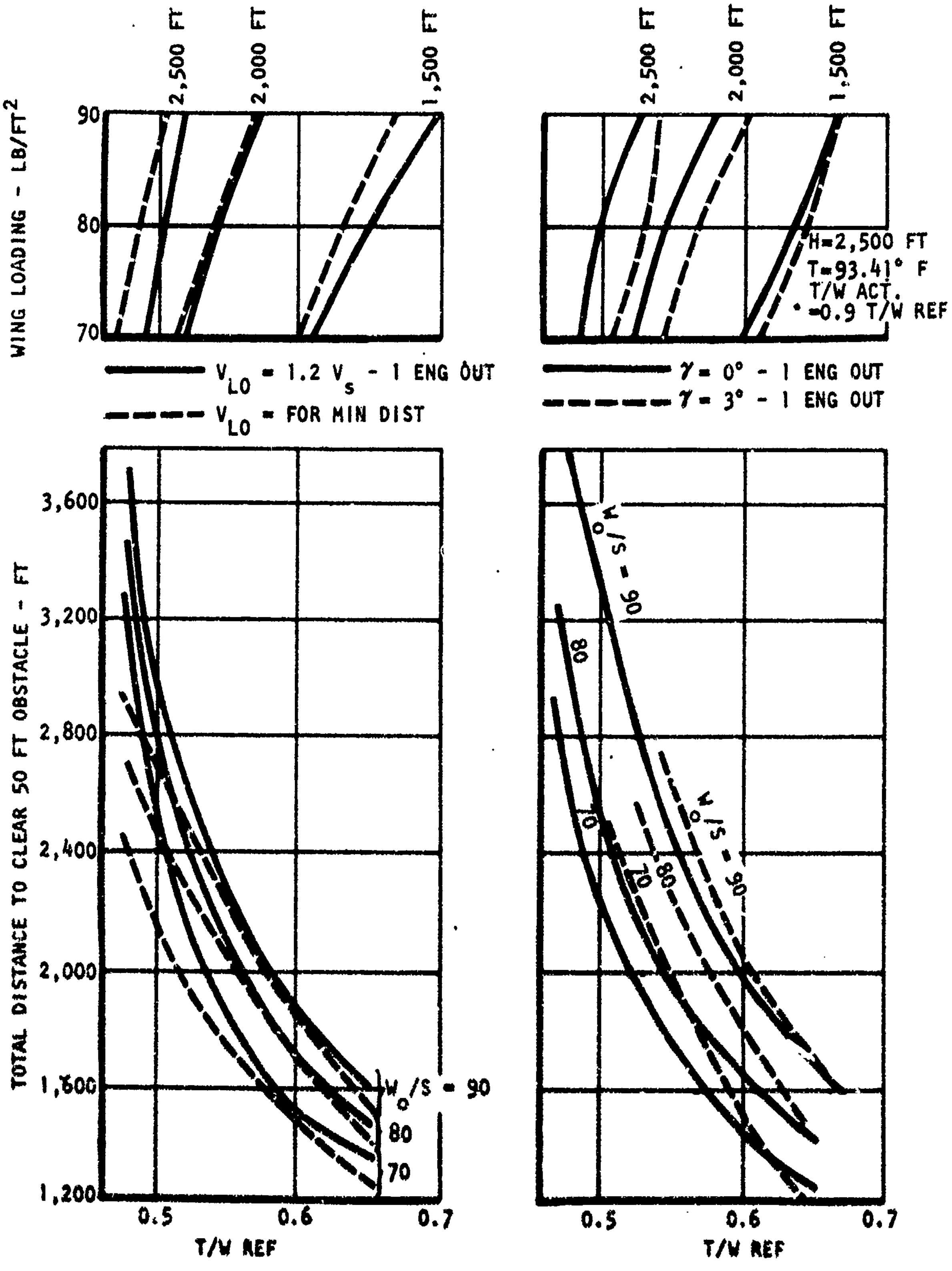


Figure 13. Summary T/W - W_o/S Effects on Takeoff Distance - δ_{f1}/δ_{f2} = 20°/40°

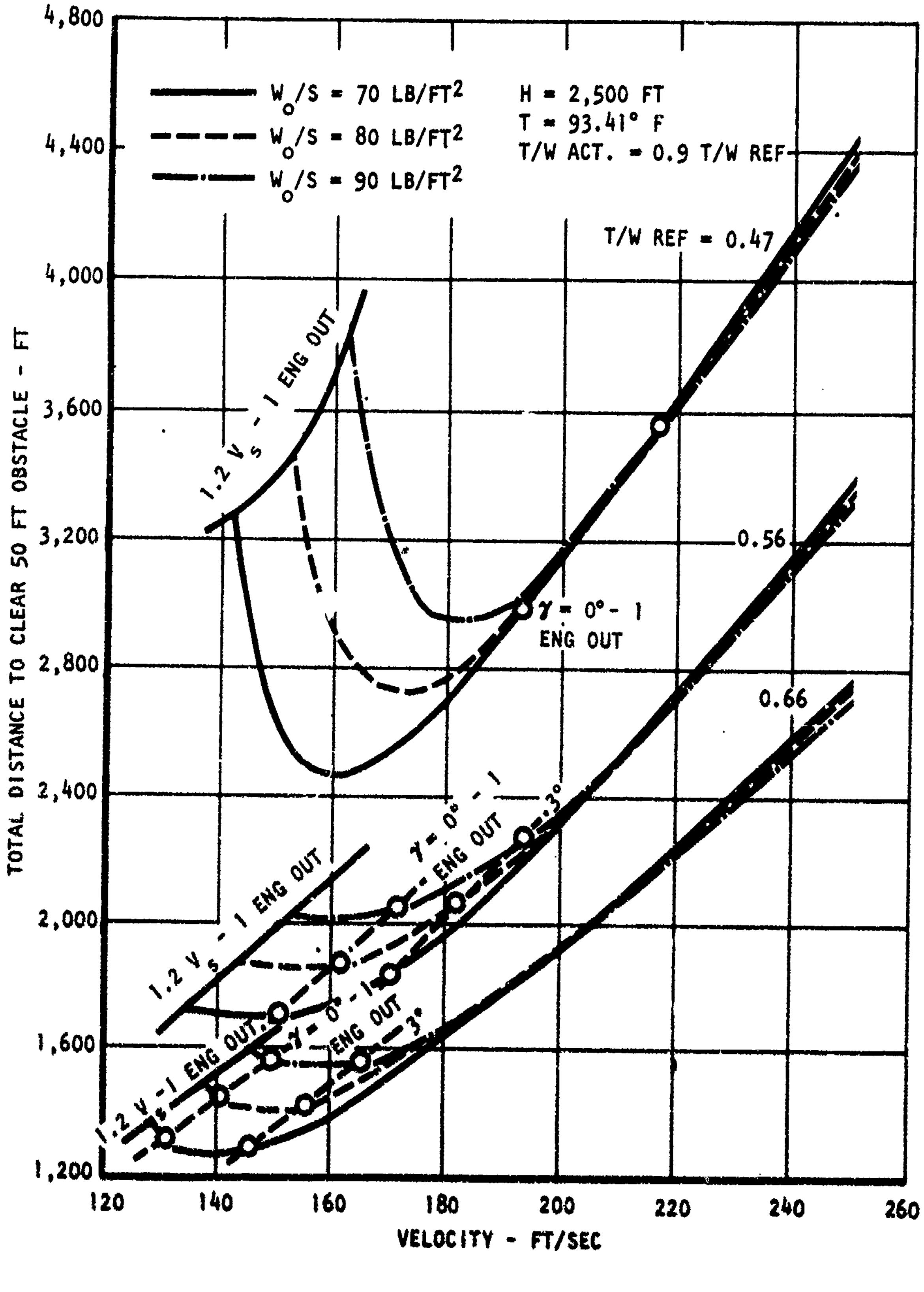


Figure 14. Total Distance to Clear 50-Foot Obstacle - δ_{f1}/δ_{f2} = 20°/40°

8. C_L at lift-off $\leq 0.9\, C_{L_{max}}$ with one engine out in ground effect

The stall velocity calculations were made with one engine out and an 8-percent lift reduction for aircraft trim. (No engine overspeed was included during this preliminary study.)

Takeoff calculations were made with a constant thrust-to-weight ratio during the entire maneuver. In addition, flap settings were held constant during any one takeoff maneuver calculation. During the ground roll, the thrust angle was deflected downward 15 degrees with an instantaneous change to 15 degrees upward at lift-off velocity. The climbout was then evaluated in this configuration.

The aerodynamic data defining lift, drag, and thrust coefficients were based on Langley Working Paper LWP 812, "Wind Tunnel Investigation of a High Thrust-Weight Ratio Jet Transport Aircraft with an External Flow Jet Flap," dated 16 October 1969. The data were used with no corrections and the basis of the intermediate flap setting of 10/20 degrees and 15/30 degrees were the results of interpolation of data shown with a tail incidence of zero degrees and an elevator deflection of -50 degrees. These data approximately represent a trimmed configuration at the CG position of the model (0.446C) which corresponds closely to that estimated for the types of vehicles under consideration. Similarly, the effects of thrust lapse with speed were neglected during the ground roll portion of the takeoff maneuver. The air distance calculations include the thrust lapse inherent in the data as presented in LWP 812 test report.

Results of this parametric study indicate that the total takeoff distance is influenced primarily by thrust-to-weight ratio with wing loading being secondary. The overriding criteria that drives the thrust-to-weight ratio required is the obstacle climb gradient with one engine inoperative. This ground rule is almost entirely independent of the flap settings investigated.

Utilizing the parametric results, an aircraft having a thrust-to-weight of 0.59 reference and a wing loading of 82 pounds per square foot will satisfy the 2,000-foot takeoff requirements. The reference thrust definition is sea-level standard day installed thrust levels.

2.7.2 LANDING STUDY RESULTS

Results of the landing performance study of the D-516-2 configuration are shown in figure 15. Total distance over a 50-foot obstacle is shown as a function of landing weight and for several breaking coefficients. Flap setting is 30/60 degrees, the altitude is 2,500 feet, and the temperature is 93.41° F.

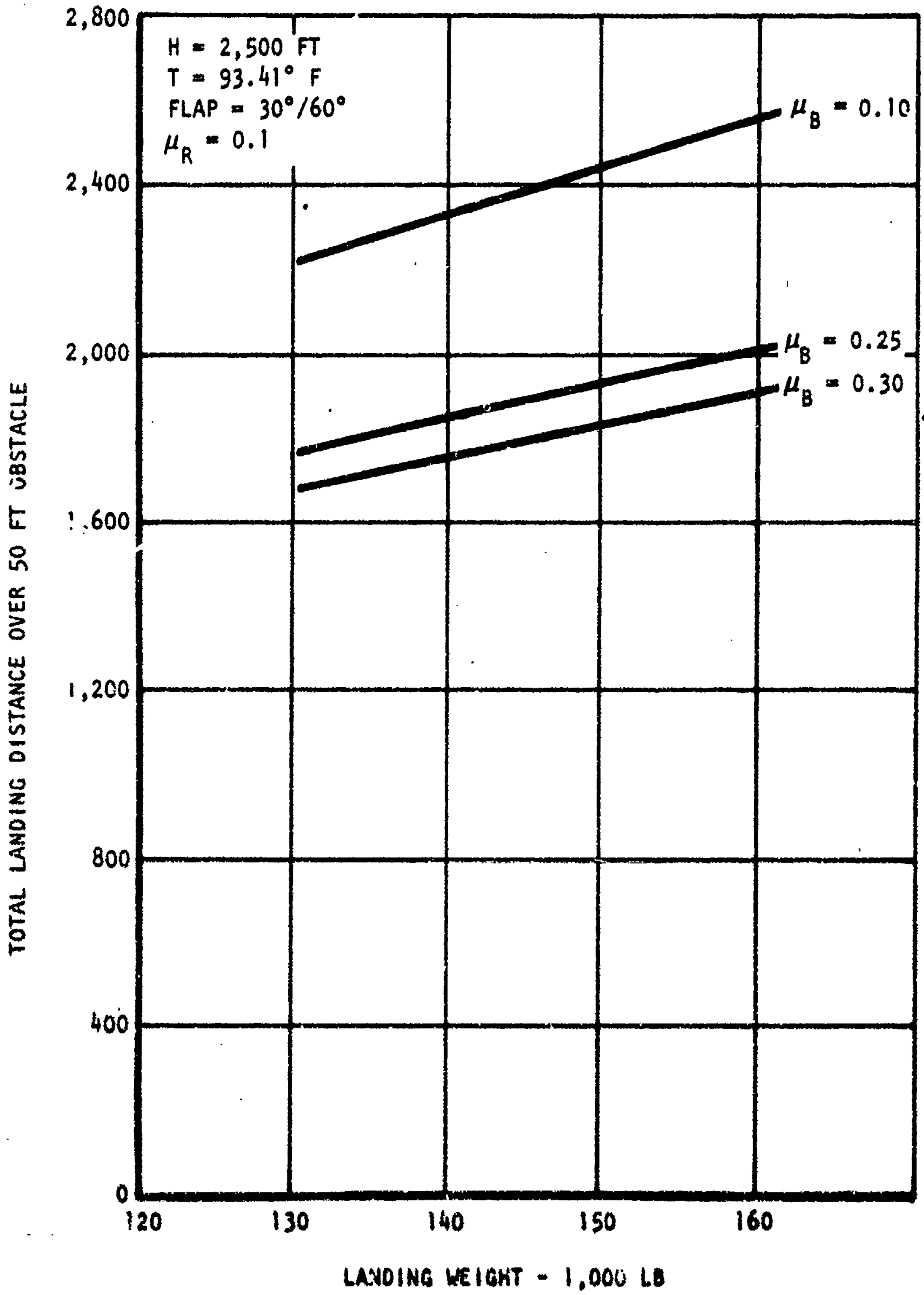


Figure 15. Landing Distance Over 50-Foot Obstacle

Landing criteria are given in paragraph 2.4. Approach is at a rate of sink of 10 feet per second (two-thirds landing gear design sink speed); a no-flare flight path is assumed; touchdown is followed by a 2-second ground roll with a rolling friction coefficient of 0.1; and, in the remaining distance, antiskid brakes are used and reverse thrust on two engines only.

The landing weight is 138,425 pounds and, at this weight, the 2,000-foot landing distance is achieved with a braking coefficient of about 0.2.

2.7.3 MISSION PERFORMANCE

Mission performance was estimated for the employment and deployment missions as shown in figures 3 and 4. The performance estimates were conducted for the D-516-2 configuration using four 90 percent size GE 13/F4B engines.

Results of the 500-nautical-mile-radius mission are shown in table VI. The fuel required for the 500-nautical-mile-mission radius is 26,545 pounds, and the weight analysis shows that this amount of fuel is available with 28,000-pound payload and a takeoff gross weight (TOGW) of 150,000 pounds.

Increased payload or range can be achieved by overloading the airplane and operating at reduced load factors. Figure 16 shows the radius that can be achieved with a 28,000-pound payload by overloading the airplane with increased fuel. Payload-range capability is given in figure 17 for several TOGW's. This chart indicates that the maximum payload that can be carried 2,600 nautical miles at design gross weight of 150,000 pounds is about 10,800 pounds; however, by operating at reduced load factor, a payload of over 30,000 pounds can be carried 2,600 nautical miles.

The basepoint, as drawn, contains 45,260 pounds of fuel, and with this fuel and zero payload, TOGW would be 140,715 pounds and ferry range would be 2,850 nautical miles. However, revisions to the wing design to move the rear spar to 61 percent of the chord and modify the wing center section box will result in a maximum fuel capacity of 56,000 pounds required for the 3,600-nautical-mile ferry mission at zero pyaload. At zero payload and 56,000 pounds of fuel, TOGW will be 151,455 pounds.

2.7.3.1 Special Payloads

In addition to the design payload of 28,000 pounds, there is an interest in carrying 44,000 pounds on an "infrequent but not one-time basis" and a 58,000-pound payload once in the life of the airplane. The airplane range capability with various payloads is shown in figure 18 as a function of TOGW. To carry 58,000 pound howitzer would require a fixed weight increase in the

TABLE VI. FUEL REQUIRED FOR 500-NAUTICAL-MILE RADIUS, BASELINE CONFIGURATION D-516-2

Leg No.	Operation	Weight (lb)	Altitude (ft)	Mach No.	Fuel Used (lb)	Time (min)	Total Time (min)	Leg Range (n mi)	Total Range (n mi)
	Initial weight	150,000.0							
1	Start and takeoff	148,201.0	0.0	0.180	1,799.0	5.000	5.000	0.0	0.0
2	Climb	144,256.1	43,241.6	0.750	3,944.9	17.687	22.687	108.57	108.57
3	Cruise	138,424.7	44,095.5	0.750	5,831.4	54.709	77.396	391.43	500.00
4	Start and takeoff	136,625.7	0.0	0.180	1,799.0	5.000	82.396	0.0	500.00
5	Climb	132,900.1	44,919.6	0.750	3,725.6	17.484	99.879	107.95	607.95
6	Cruise	127,516.2	45,798.0	0.750	5,383.8	54.802	154.681	392.06	1,000.00
7	Loiter	124,782.3	0.0	0.300	2,734.0	30.000	184.681	0.0	1,000.00
8	Reserve	124,782.3	0.0	0.0	1,327.2	0.0	184.681	0.0	1,000.00
Total fuel = 26,545 lb									

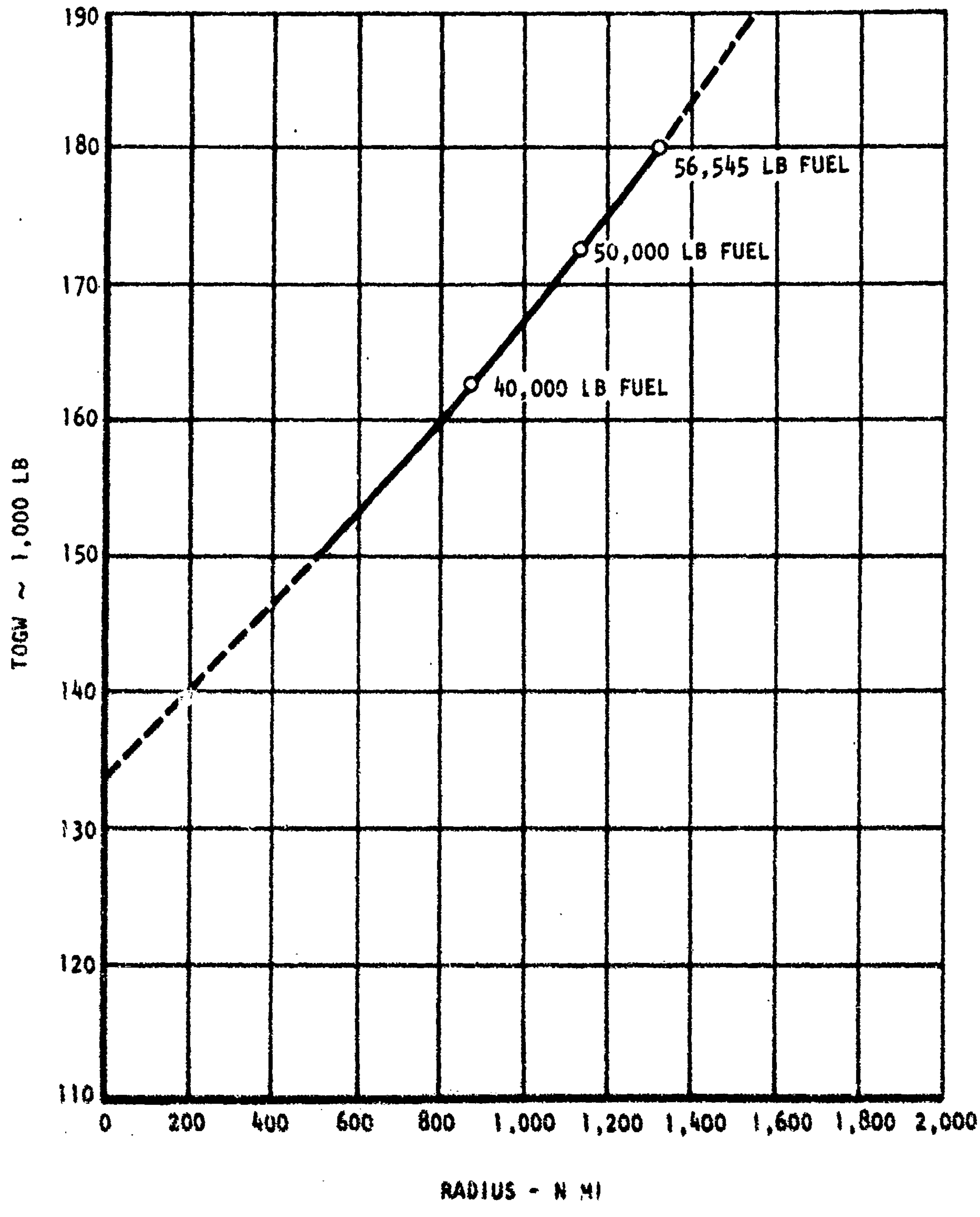


Figure 16. TOGW Versus Radius - Payload = 28,000 Pounds

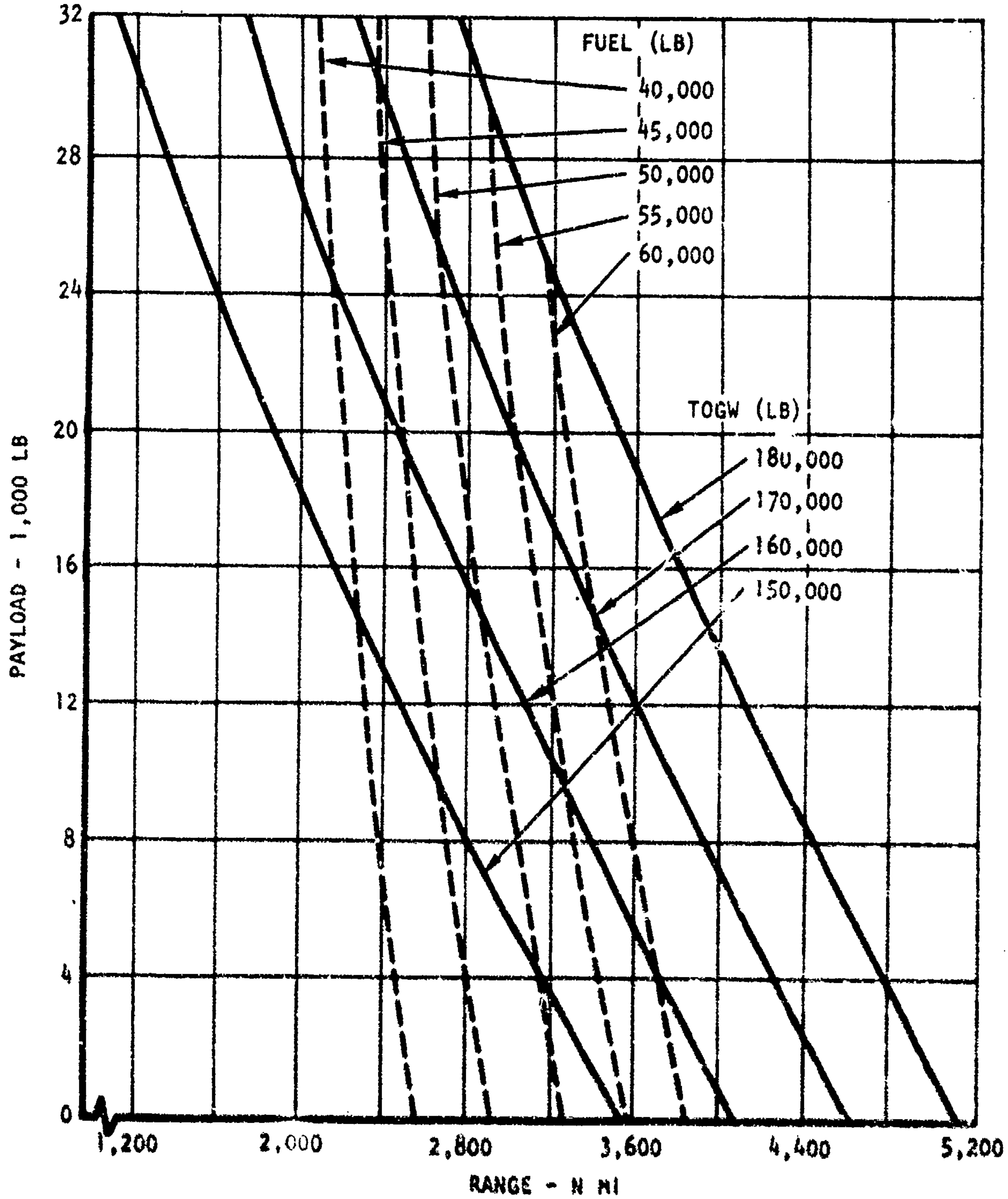


Figure 17. Payload Versus Range

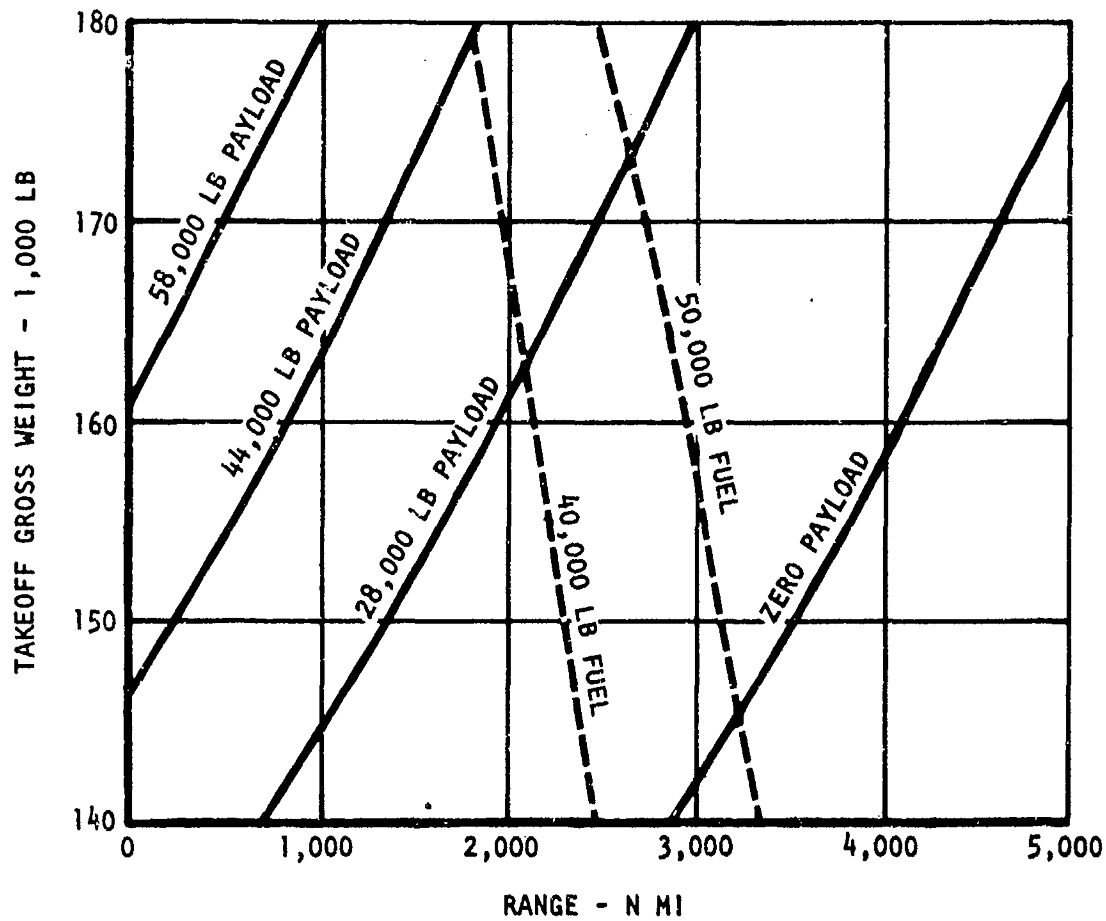


Figure 18. Takeoff Gross Weight Versus Range for Various Payloads

landing gear carry-through structure of 75 pounds at TOGW of 160,000 pounds and 100 pounds at a TOGW of 170,000 pounds. These values, multiplied by the growth factor of 2.725, would result in a TOGW increment of 204 pounds for a range capability of about 100 nautical miles and 273 pounds for a range capability of about 700 nautical miles.

2.8 TRADE STUDY SUMMARY

The STOL TAI design requirements, as shown in table I, indicate interest in the following trade studies:

Trade	Basepoint Design	Trade Value
Mission radius (n mi)	500	750
Cruise speed (M)	0.75	0.85
Takeoff distance (ft)	2,000	±500
Cargo bay size (ft)	12 x 12 x 45	12 x 12 x 55
Sea-level penetration speed (knots)	-	400

The trade studies conducted originate about the D-516-2 configuration with appropriate drag modification and vehicle scaling pertaining to the details of the specific trade. Reference TOGW for the D-516-2 configuration is 150,000 pounds with 26,535 pounds of fuel. This vehicle has for its propulsion system four 90 percent size GE13/F4R 7.8 BPR turbofans engines. Trade study solutions are shown in figures 19 through 23.

2.8.1 WEIGHT GROWTH

The weight growth trade establishes the dead weight effects on aircraft TOGW. The D-516-2 configuration is the trade basepoint about which the fuel fraction available data was generated. Dead weight effects were produced by arbitrarily adding or subtracting weight from the vehicle fixed load. Sizing criterion were the 500-nautical-mile design mission and the 2,500 feet, 93.4° F day takeoff conditions. Takeoff and landing requirements were satisfied by a midmission thrust-to-weight ratio of 0.53 and a 82.0-pounds-per-square-foot wing loading. The results of this study indicate that, with a 10,000-pound dead weight removal, the gross weight reduces by 26,500 pounds. With a 10,000-pound dead weight addition, the gross weight increased by 28,000 pounds, and a 20,000-pound dead weight increase results in a 59,000-pound TOGW increase. Figure 19 depicts the weight growth.

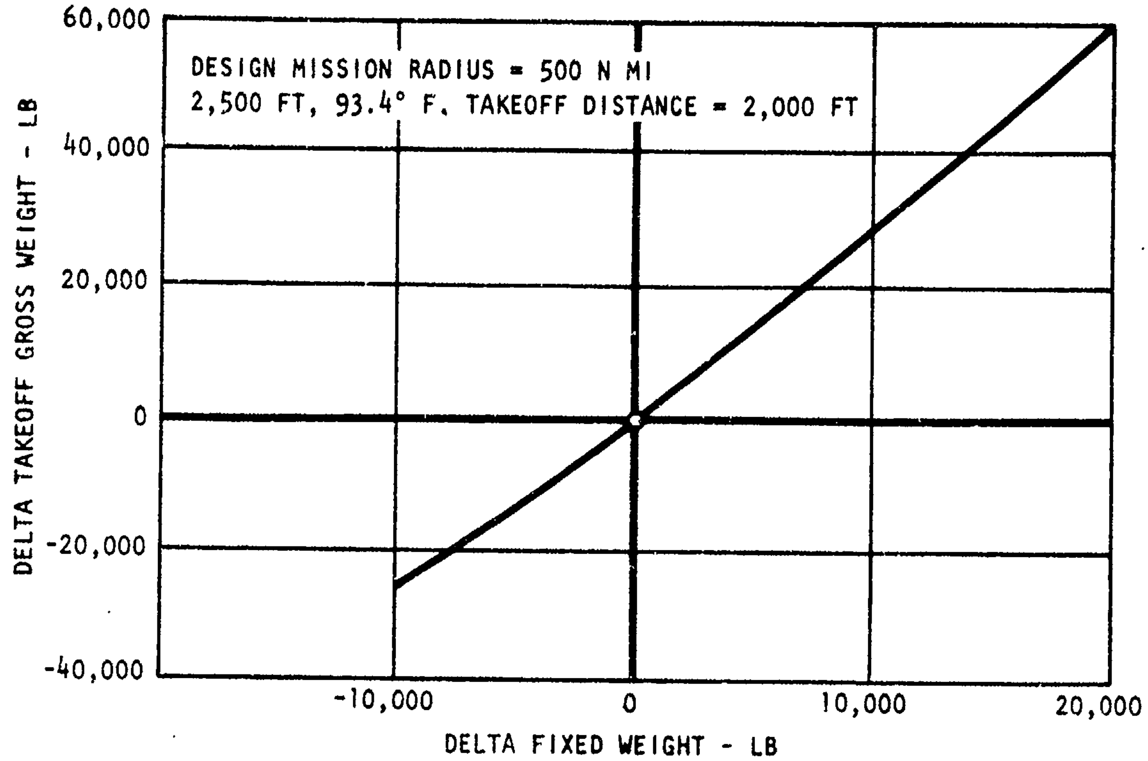


Figure 19. Weight Growth Factor

2.8.2 DESIGN MISSION RADIUS TRADE

This trade involves increasing the design mission radius from 500 to 750 nautical miles, and maintaining 2,000-foot takeoff and landing distance performance. The takeoff and landing performance are satisfied by maintaining thrust-to-weight of 0.53 and 82-pounds-per-square-foot wing loading at the mismission weight. The midmission weight used in sizing the wing and engines was based on TOGW minus approximately 45.5 percent of the fuel required. The result of this trade is shown in figure 20 which indicates an aircraft with a TOGW of 172,750 pounds or 24,750 pounds delta gross weight would satisfy the 750-nautical-mile mission radius. This aircraft would have a 21.7 percent fuel fraction (37,487 pounds of fuel), a 1,905-square-foot wing area, and 102 percent GE13/F4B engines. Other wing geometry would remain the same as the D-516-2 baseline configuration.

2.8.3 CRUISE SPEED TRADE

The cruise speed trade involves major changes to the aircraft wing and engines. The objectives of this trade were to establish the gross weight and aircraft geometries that would allow efficient cruising up to and including 0.85 mach number. This trade was done in two steps from the D-516-2 basepoint. The first step was a design that would have a 0.82 mach cruise speed.

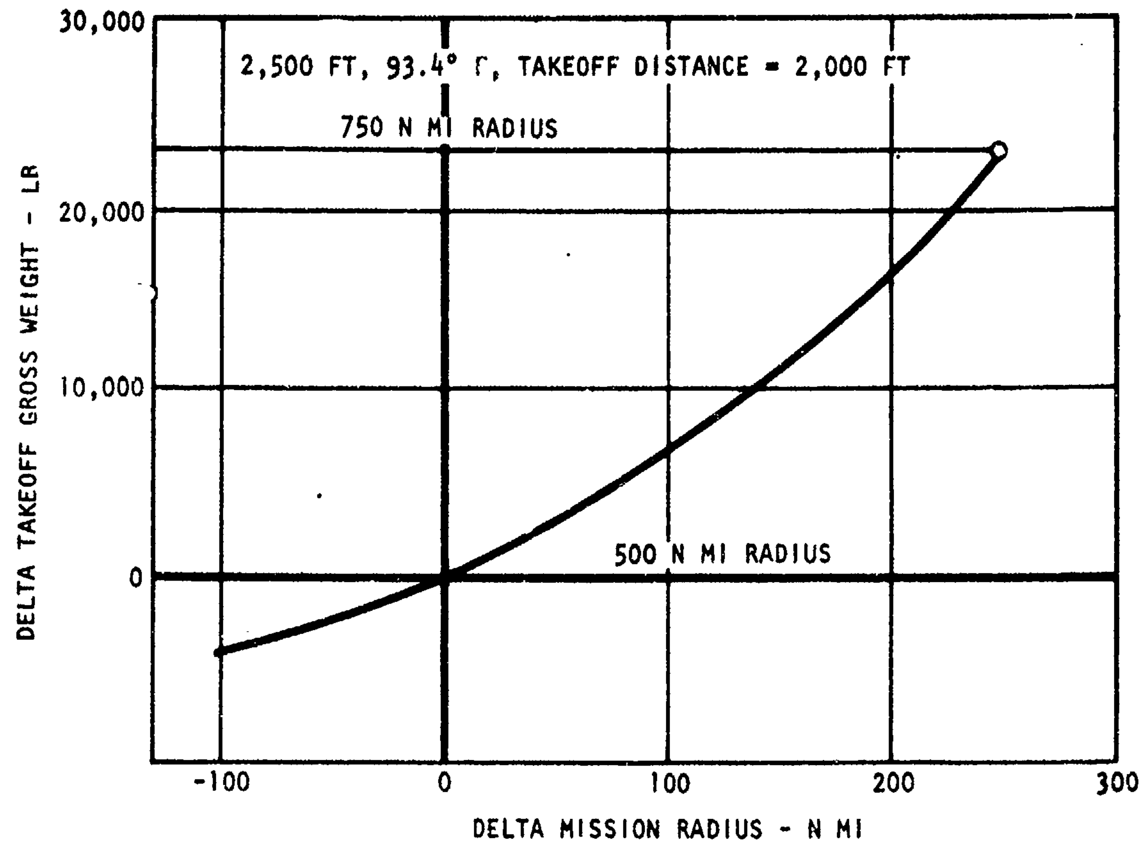


Figure 20. Design Mission Radius Trade

The engine selected was a reduced bypass ratio (BPR), from 7.8 to 5.0, which was more compatible for the higher cruise speed. The GE13/F4B has a 7.8 BPR, whereas the GE13/F3A has a 5.0 BPR. The 5.0 BPR GE13/F3A engine was used for both speed conditions investigated. Engine weight scaling was based on engine size with a 1.25 exponent. The engine sizing criteria were the takeoff and landing conditions at the 2,500-foot, 93.4° F day conditions. A thrust-to-weight ratio of 0.56 was required for the 0.82M design vehicle to maintain the 2,000-foot field length. The increase in thrust-to-weight ratio results from the loss of lift due to sweeping the wing to improve aircraft drag divergence. An increase in engine size was the most efficient means to reduce takeoff distance. Using an increase in wing area (lower wing loading) would result in an aircraft that would be approximately 600 to 700 pounds heavier than one where engine size was used to regain the takeoff performance. A 0.57 thrust-to-weight ratio is required for 0.85M design for the 2,000-foot takeoff requirement. For both designs, an 82-pound-per-square-foot wing loading was used when determining engine size.

Wing geometry selection criteria were based on characteristics to increase drag divergence mach number and retain good low-speed aerodynamic characteristics. A 10-percent wing was selected as the minimum thickness from a practical design point of view. A 2-percent decrease in thickness increases drag divergence by 0.03 mach number. The remaining drag divergence mach number increase was accomplished by increasing quarter chord sweep for the 0.82M design to 34 degrees and to 39 degrees for the 0.85M design. An aspect ratio reduction from 8.0 to 7.0 was made to offset weight penalties associated with reduced thickness. These geometric changes, along with the reduced BPR engine, produce the cruise speed trade shown in figure 21. The vehicle that satisfies the 0.82M cruise would have a TOGW of 168,500 pounds, a fuel fraction of 18.65 percent (31,425 pounds of fuel), a 1,855-square-foot wing area, and a 121.7-percent size GE 13/F3A engines. For 0.85M cruise design, the TOGW is 178,500 pounds, fuel fraction 18.7 percent (33,380 pounds fuel), 1,995-square-foot wing area, and 131.8 percent size GE13/F3A engine.

A check was made to determine fuel required at zero payload to fly the 3,600-nautical-mile ferry mission at a 0.85M cruise speed. The fuel required for this mission is 68,180 pounds. The TOGW would be 185,300 pounds which is a 9,200-pound overload for the airplane designed at a normal TOGW of 178,500

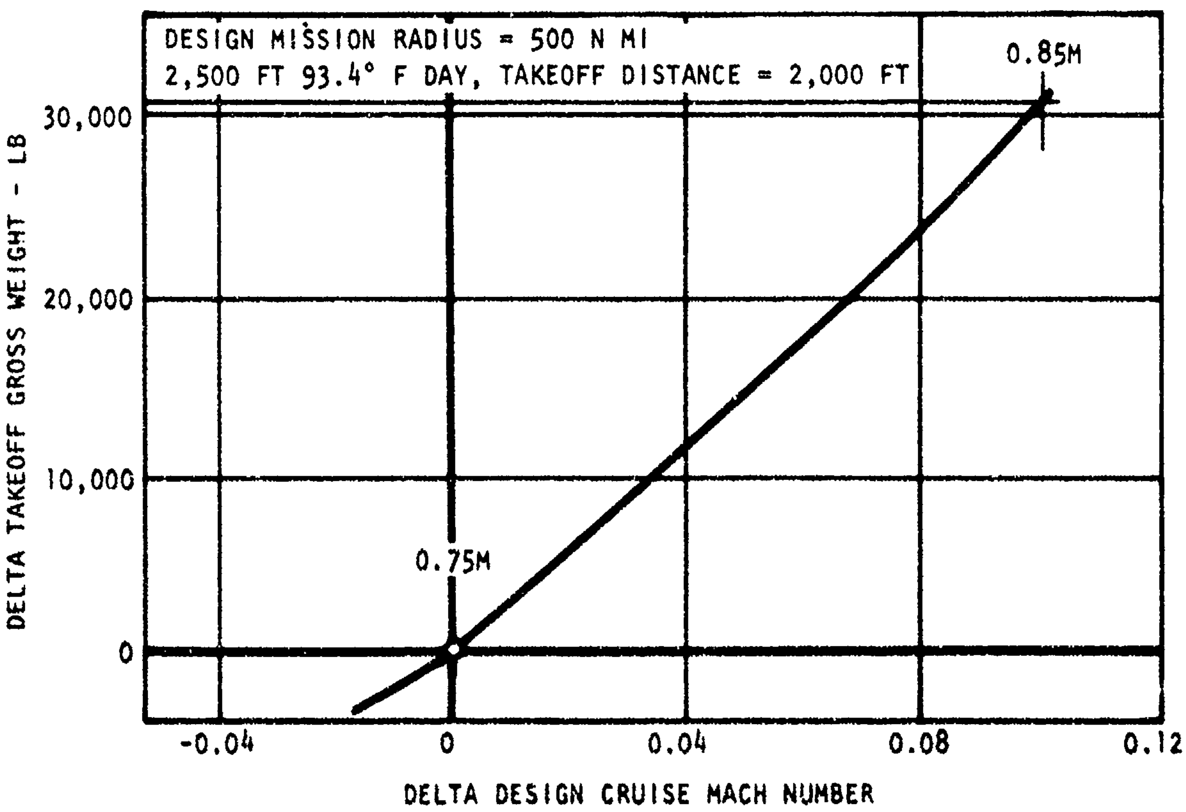


Figure 21. Design Mission Cruise Speed Trade

pounds. A layout was made to determine fuel capacity of the 1,995-square-foot wing, which indicated 62,000 pounds capacity. Minor variations of spar locations and center section shaping should provide the required 68,180-pound capacity.

2.8.4 TAKEOFF FIELD LENGTH TRADE

The takeoff field length trade influences the aircraft design primarily through the engine sizes required for the takeoff and landing distances of interest. This trade was initiated about the D-516-2 configuration with appropriate component weight scaling and with engine size determined by field length at an 82-pounds-per-square-foot wing loading. Results of this study are shown in figure 22. For the 2,500-foot takeoff distance, a thrust-to-weight ratio of 0.50 is required, and the 1,500-foot requirement is 0.67 at the 2,500-foot 92.4° F day condition. The results of this trade indicate that a 500-foot takeoff distance increase would reduce TOGW by 6,000 pounds, and a 500-foot takeoff distance reduction would produce 27,500 pounds weight increase.

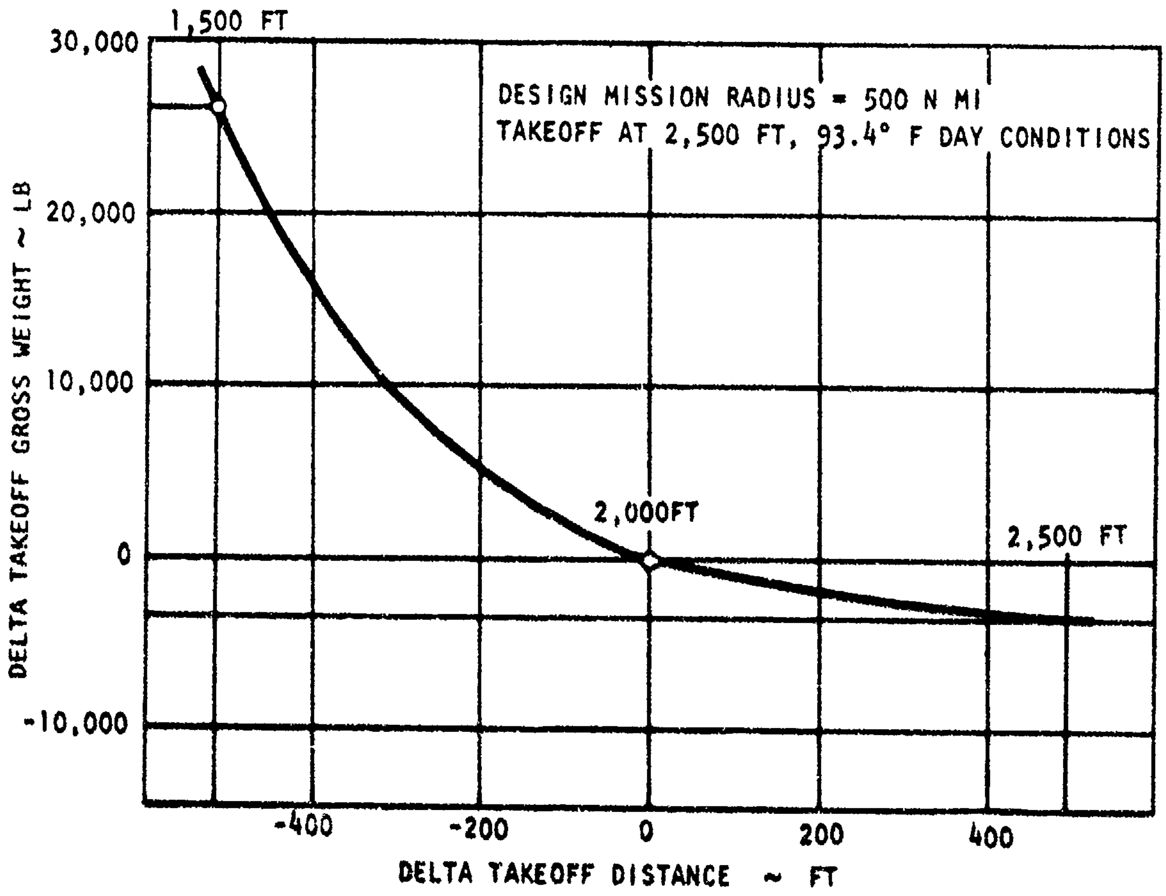


Figure 22. Takeoff Distance Trade

2.8.5 CARGO BAY LENGTH TRADE

The effect of increasing the aircraft cargo bay from 45 feet to 55 feet was determined by increasing the fuselage length with addition of a constant-diameter section and relocating the wings and tails. It was assumed that rearrangement of the aircraft components would be accomplished such that the major effect would be directly associated with the geometry of the fuselage alone. These modifications to the fuselage results in a 12-percent wetted area increase, a 1,765-pound basic fuselage weight increase, and a 500-pound weight increment to furnishings. These changes were incorporated in the basic D-516-2 configuration, and the aircraft was sized to meet the 500-nautical-mile radius and the 2,000-foot takeoff and landing criteria. The results of this trade are shown in figure 23 which indicates that an aircraft weighing 157,500 pounds with a 17.75-percent fuel fraction (27,601 pounds of fuel), a 1,755-square-foot wing area, and 94.1 percent GE13/F4B would satisfy the design requirements. The takeoff weight increment for the 55-foot bay is 7,500 pounds.

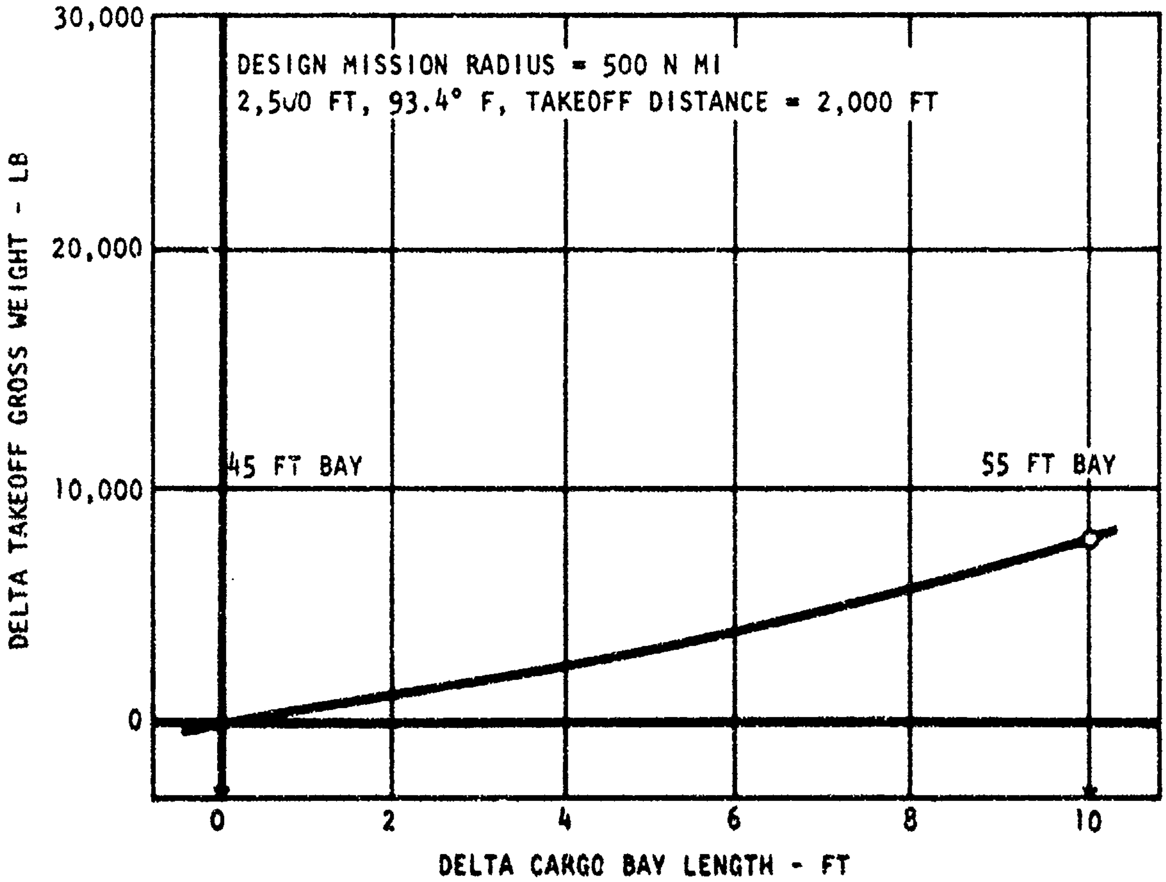


Figure 23. Cargo Bay Length Trade

2.8.6 SEA LEVEL PENETRATION SPEED

The airplane design q was established by 0.75M cruise speed capability at 20,000 feet, and the maximum cruise speed below 20,000 feet was limited by the design q. This results in a sea-level cruise speed of 311 knots for the basepoint airplane. A study was conducted to determine the fixed-weight penalty of the basepoint airplane to increase the sea level cruise speed to 400 knots. The weight increment is 495 pounds, and this value times the growth factor of 2.73 would result in a TOGW increment of 1,250 pounds to maintain constant airplane performance.

SECTION III

DESIGN CRITERIA

The design criteria applicable to the refined baseline configuration are similar to those established during the part I study phase. The development of the refined configuration was initiated about the configuration, as determined by those design requirements as outlined in paragraphs 2.2 through 2.4. Modifications were made where a particular design rule was changed. Vehicle sizing for the refined configuration adheres to the design criteria as discussed in this section.

3.1 DESIGN REQUIREMENTS

The STOL-TAI design requirements are shown in table VII. These requirements are similiar to those presented in paragraph 2.2, with the exception of the takeoff and landing performance. Takeoff and landing performance criteria were revised during the early portion of the part II study and are presented in paragraph 3.2. Another change involves the use of a new engine that is based on an existing cycle used in the F101 engine program. This engine is consistent with the growth version of General Electric's GE13/F10 engine.

3.2 STOL TAKEOFF AND LANDING CRITERIA

The revised STOL takeoff and landing rules for the STAI programs that are applicable for part II are described in this section. These rules are the criteria used in the update of the baseline configuration. The STOL maneuvers are evaluated at midmission weight, as determined by the 500-nautical-mile radius mission with 28,000-pound payload onboard. The rules as specified provide for a 3-degree climb gradient at a minimum velocity. Lift-off velocity limiting values are controlled by the climb gradient requirement, as well as those imposed by the maneuver margins. The takeoff and landing ground rules as specified by AFFDL are shown in tables VIII through X.

3.3 MISSION DESCRIPTIONS

The STOL-TAI missions are the 500-nautical-mile radius design mission and 2,600-nautical-mile range mission. (Refer to paragraph 2.3.) The design mission payload is 28,000 pounds retained throughout the entire mission. STOL takeoff and landing weights are determined by the midpoint weight just after the high-altitude optimum mach cruise. This weight is used to determine the STOL takeoff and landing performance. Vehicle sizing involving takeoff and landing performance is based on the ground rules discussed in paragraph 3.2 and the midmission weight definition.

TABLE VII. MST DESIGN REQUIREMENTS

	Basepoint	Trade-offs
Missions		
Design radius	500 n mi	750 n mi
Payload (both ways)	28,000 lb	
Maneuvering limit	3 g	
Unrefueled range (2.5 g) will be 2,600 n mi with reduced cargo, and 3,600 n mi with no cargo.		
Performance		
Cruise speed	0.75 M at h ≥ 20,000 ft	0.85 M
Penetration speed at SL for 50 n mi	-	400 kt
(This requirement refers to a terrain-following environment with associated avionics. This is not a baseline, only a trade-off.)		
T.O. balanced field length and landing distance over 50 ft 2,500 ft alt/93° F	2,000 ft	+ 500 ft
Design Characteristics		
Max design TOGW	160,000 lb	
Bay size (minimum length)	12 x 12 x 45 ft	
Crew: pilot, copilot, navigator, loadmaster		
Propulsion: growth version of existing-type engine		
Miscellaneous features		
Aerial refueling		
Pressurized crew and cargo compartments		
Wheel drive for ground handling at remote site		
Landing gear for CBR-6 allowing for 200 passes and lower CBR at reduced payload. Design sink speed of 15 fps.		
Antiskid brakes		
Minimum AGE use and no special facilties		
Vulnerability/survivability; maximum protection and fail-safety		
Satisfactory engine-out control		
Maximum evasive maneuverability		
Paratroop and equipment airdrop capability. Equipment airdrop up to 400 kt.		
463L compatibility		
Self-contained off-load capability		
Drive-on loading capability (not drive-thru)		
24 hr continuous operations		
115 Pndb at 500 ft sideline		
3-engine takeoff capability from assault strip with sufficient fuel for 500 n mi return leg, without cargo		

TABLE VIII. TAKEOFF RULES

Field Length Definition	Distance from brake release to lift-off or stop when one engine fails at the most critical point along the path (engine-out balanced)
Rolling coefficient of friction	$\mu_R = 0.1$, $\mu_{BR} = 0.3$
Rotation rate	$\dot{\theta} \leq 8°$ per second
Speed limitations	$V_F \geq V_{mcg}$ (OEI)
	$V_{LO} \geq 1.05\ V_{mca}$ (OEI, IGE)
	$V_{CO} \geq 1.10\ V_{mca}$ (OEI, OGE)
Maneuver margins	At V_{LO}: ≥ 1.10 g (OEI, IGE)
	At V_{CO}: ≥ 1.30 g (AEO, OGE)
Climb requirement	$\geq 3°$ (OEI)
Rejected takeoff factors	Engine failure recognition time = 1 sec. Time delay to deploy deceleration devices = 2 sec.

TABLE IX. LANDING RULES

Field length definition	Distance to stop from 50 ft threshold height
Flightpath	Design R/S $\leq$ 1,000 fpm
	R/S $\leq$ 2/3 gear design R/S
	No flare
Speed limitations	$V_{TH} \geq 1.1 V_{mca}$ (OEI)
	$V_{TD} \geq 1.1 V_{mtd}$ (OEI, IGE)
Maneuver margins	At V_{TH}: $\geq$ 1.30 g (OEI)
	At V_{TD}: $\geq$ 1.15 g (OEI, IGE)
Stopping rules	Time delay to deploy deceleration devices = 2 sec
	$\mu_{BR} = 0.3$
	Symmetrical reverse thrust: $\leq$50% of two-engine thrust for four-engine designs
Waveoff requirement	From 50 ft (AEO)
	From 100 ft (OEI)
	Configuration changes permitted

TABLE X. NOMENCLATURE

V_F	=	Engine failure speed. The actual speed at which the failure occurs.
V_{mcg}	=	Ground minimum control speed. The minimum speed at which controllability is demonstrated (during the takeoff run) to be adequate to permit proceeding safely with the takeoff, and using average piloting skill, when the critical powerplant is suddenly made inoperative.
V_{mca}	=	Air minimum control speed. The minimum control speed in the air with the critical powerplant inoperative. When the critical powerplant is suddenly made inoperative at this speed, it shall be possible to recover control of the airplane, with the powerplant still inoperative, and maintain it in straight flight at that speed, with a bank angle less than 5°.
V_{LO}	=	Liftoff speed
V_{TH}	=	Threshold speed or final approach speed
V_{TD}	=	Touchdown speed
V_{CO}	=	Climbout speed
V_{mtd}	=	Minimum touchdown speed. The lowest speed at which the airplane can touch down.
AEO	=	All engines operating
OEI	=	One engine inoperative
IGE	=	In ground effect
OGE	=	Out of ground effect

The alternate 2,600-nautical-mile range mission profile is performed at a takeoff gross weight corresponding to a 2.5 g load factor with a reduced payload. This mission determines the wing size through the requirement that mission fuel must be contained in the wing. In addition, the speed requirement at 20,000 feet sizes the engine at the end of the climb weight. A 3,600-nautical-mile range is desired with no cargo onboard and using fuselage bay tanks. Takeoff gross weight for this range is not to exceed that for a 2.5 g load factor.

Section IV

CONFIGURATION DEFINITION

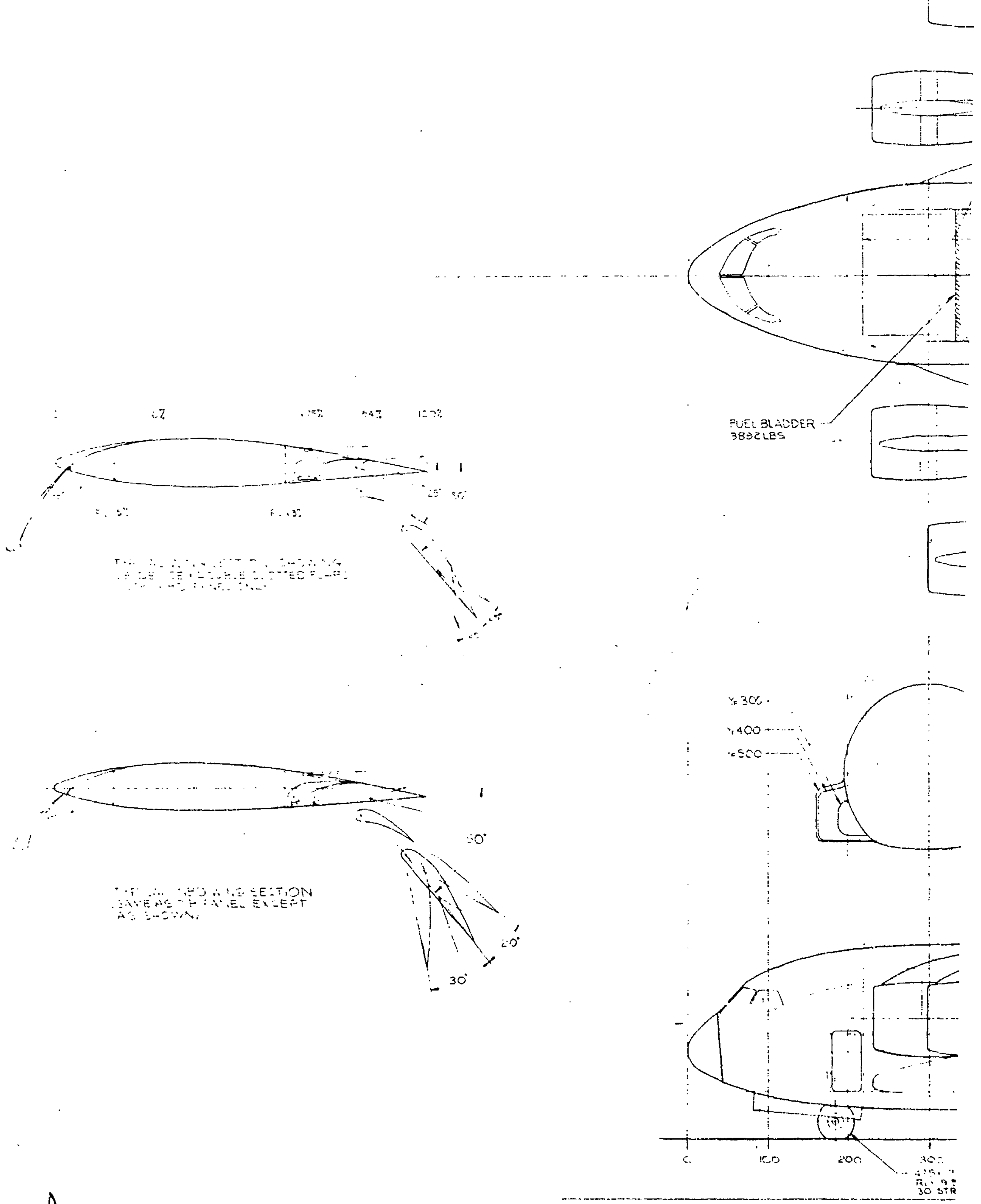
The STOL-TAI refined basepoint configuration is illustrated in figure 24 with an inboard profile shown in figure 25 and a three view in figure 26. This vehicle is an externally blown flap propulsion-lift airplane designed to the requirements contained in section III. These criteria include a 28,000-pound payload, a 12 x 11.7 x 45-foot long cargo bay, a 500-nautical-mile design mission radius, a mach 0.75 cruise speed, and a 2,000-foot STOL field at 2,500 feet, 93.4° F day conditions. The design takeoff gross weight for the basic 500-nautical-mile radius mission is 159,310 pounds. The vehicle has an initial thrust to weight ratio of 0.516 at sea level standard day condition, and a 99.6-pounds-per-square-foot wing loading. Installed midmission thrust to weight ratio is 0.545 at 2,500 feet, 93.4° F day condition with a 90.7-pounds-per-square-foot wing loading. The engine used is a 93-percent size advanced version of the GE 13/F10 engine, rated at 22,320 pounds uninstalled thrust at sea level standard day conditions. The 100-percent size engine uninstalled thrust to weight is 8.0, which is consistent with General Electric's proposed GE13/F10 growth version engine. The engine is a 6.5 bypass ratio turbofan engine, using a modified GE F101 core.

4.1 GENERAL DESCRIPTION

The STOL-TAI configuration is a high wing, T-tail arrangement using four turbofan engines. The fuselage has a 210-inch-diameter circular cross section, with some flattening underneath to provide adequate ground clearance while maintaining a 56-inch floor height above the ground. The crew compartment is sufficient for a four-man crew, rest facilities, and avionics equipment. The aft fuselage has been faired as sharply as practicable and yet avoids severe separation drag penalties. A beaver tail has been incorporated to simplify cargo bay design and permit airdrop by opening the aft door inward rather than out into the airstream. A spoiler is provided at each of the troop doors located at the aft end of the cargo bay on each side of the fuselage.

The wing selected has an 8.0 aspect ratio and a quarter chord sweep of 25 degrees. The airfoil section is 64A300 series, with a 12.5 percent exposed thickness ratio at the fuselage juncture. Midspan thickness ratio is 11.5 percent, with a 10-percent thick wingtip. The wing trailing edge has been modified from span station 367 inboard to improve wing body flow characteristics at the 0.75M design cruise speed.

The wing incorporates full-span spoilers, powered leading edge slats, and double-slotted flaps. The chord of the inboard two-thirds of the slat is 20 percent of the wing chord; this section can be extended to 30 degrees below



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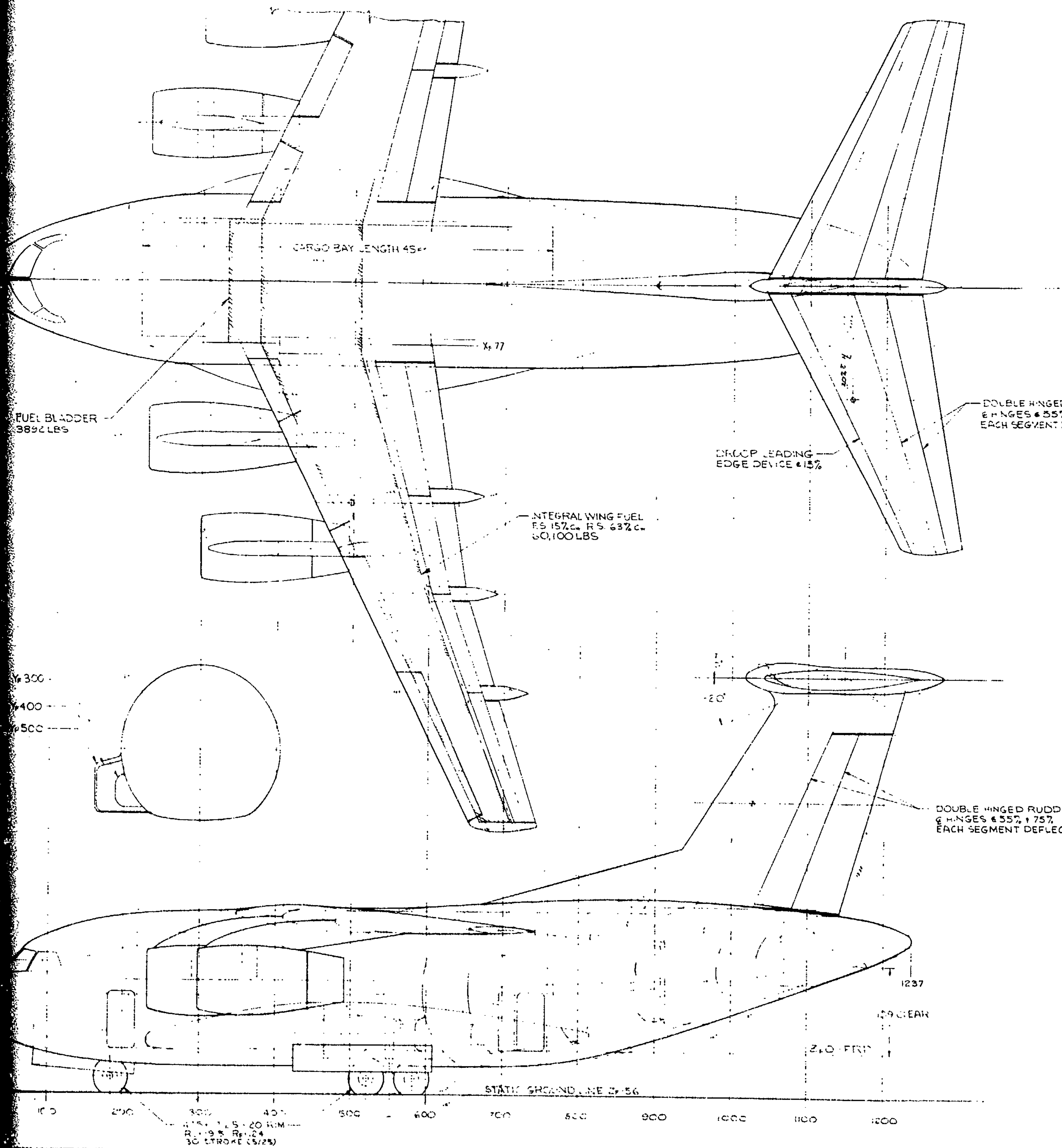

CARGO BAY LENGTH 45
X=77
FUEL BLADDER
3892 LBS
DROOP LEADING
EDGE DEVICE @ 15%
INTEGRAL WING FUEL
60,100 LBS
Y=300
Y=400
Y=500
1237
STATIC GROUND LINE Z=56
100
200
300
400
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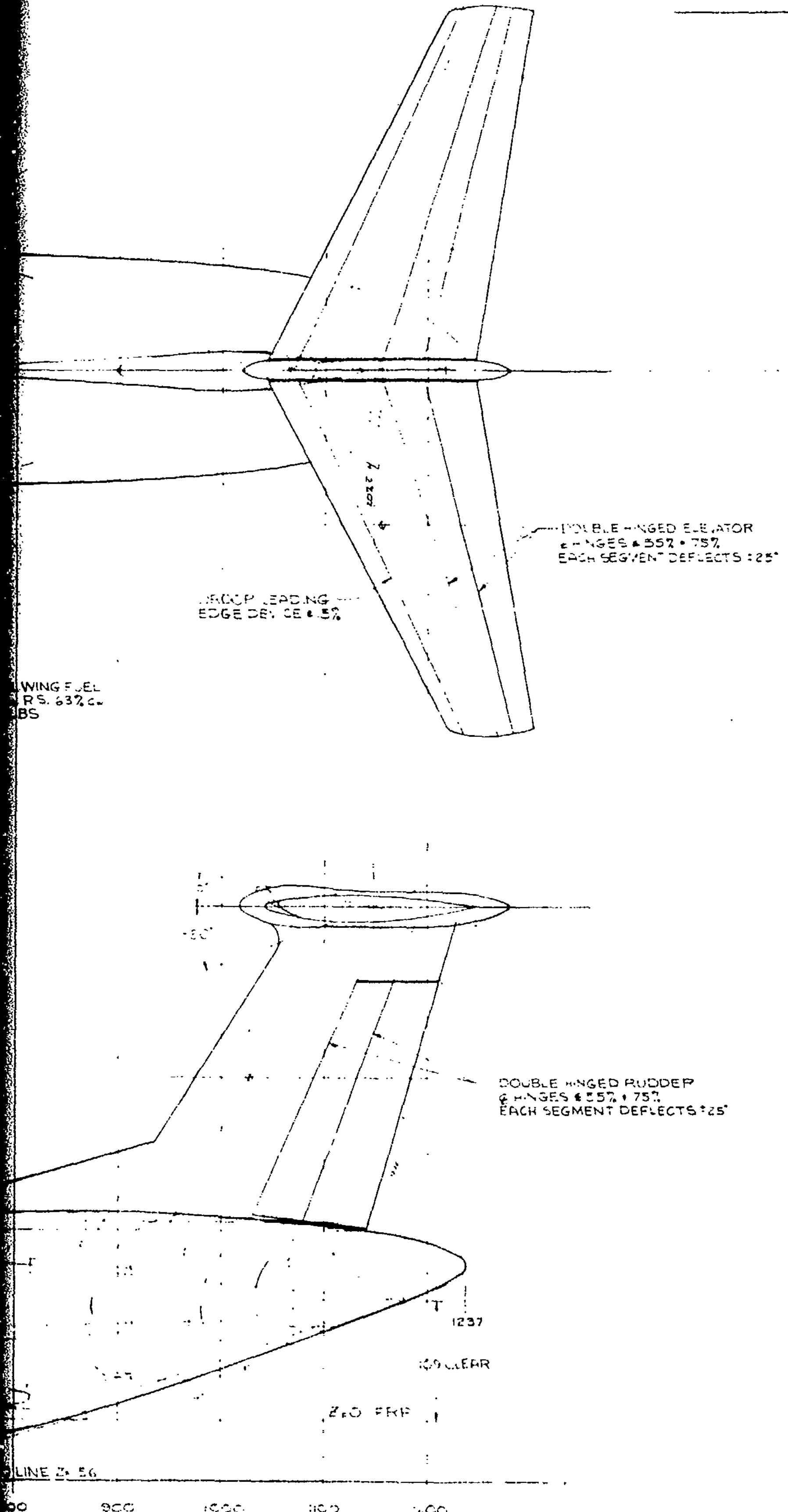
800
900
1000
1100
1200

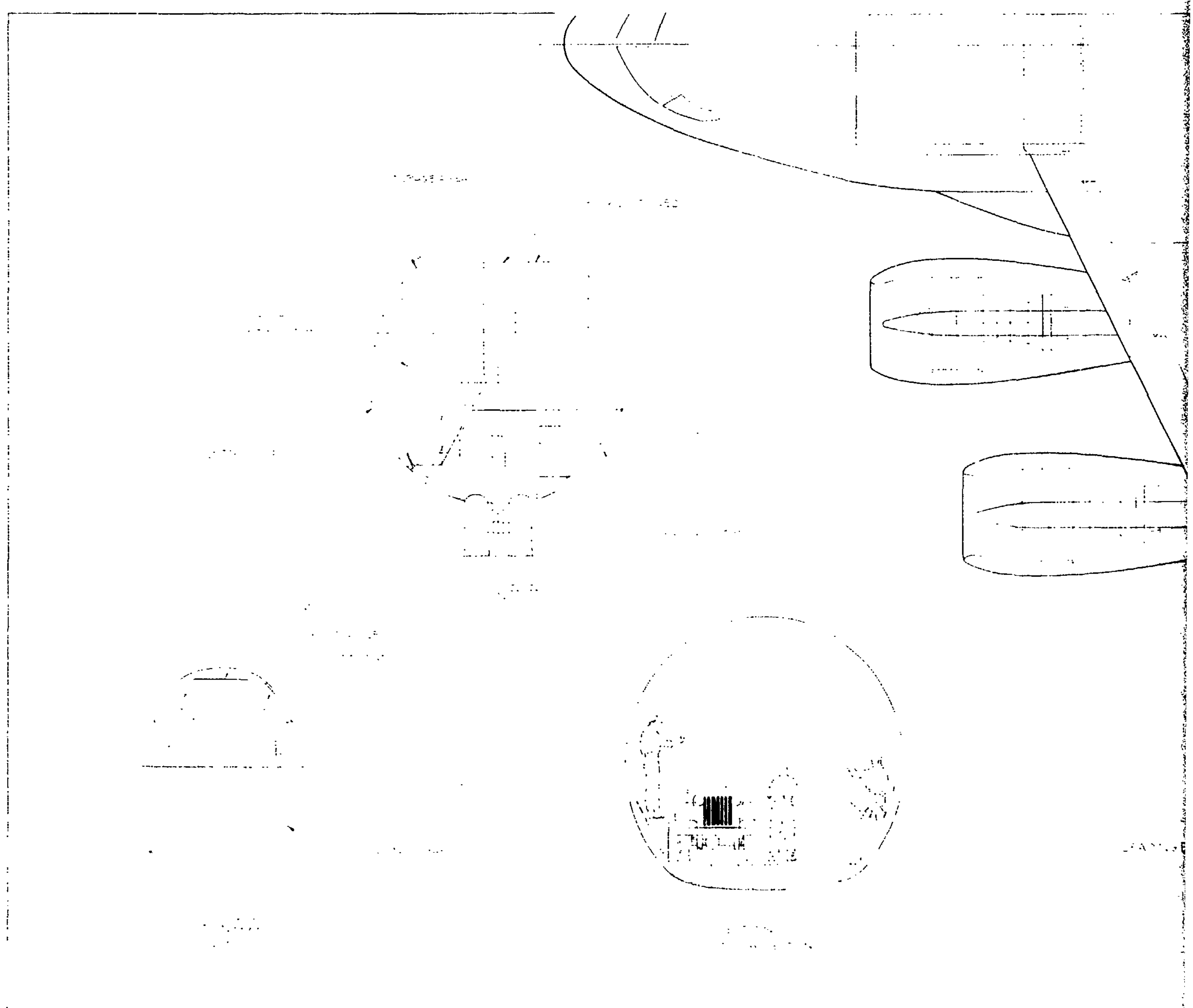


GEOMETRY				
ITEM	WING TRAP	WING TOTAL	HORIZ	VERT
$S \sim FT^2$	1430	1600	665	410
AR	8.95	8	4.50	1.40
λ	0.40	-	0.40	0.60
b	1357.57		656.44	287.50
$\Lambda_{c/4}$	25°	-	25°	30°
C_R	216.7	290	208.40	256.70
C_T	86.68		83.36	154.02
$\bar{c}$	6.	189	154.81	209.64
$\bar{X}$ OR $\bar{Z}$	291	274	140.67	131.77
DIHEDRAL	0° LOWER M.		0°	~
INCIDENCE	±4° CR → -1° CT		VARIABLE	~
AIRFOIL	64A312.5 CR EXP TO 1=265=11.2% 64A310 CTIP		64A212 CR 64A008 CT	64A012 CR 64A010 CT
ℓ	~		659	530
V	~		.45	0.100
$z/\bar{c}$	-		1.65	~

PROPULSION (4) 93% SIZE GE 13/F10 ENGINES
WITH GIMBALLED NOZZLES
± 5° DEFLECTION

TARGET TOGW: 159,310 LBS
PAYLOAD 28,000 LBS.
FUEL 33,151 LBS.
MID-MISSION WEIGHT 45,240 LBS
2600 N MI FUEL: 63,992 LBS

Figure 24. Refined Basepoint Configuration



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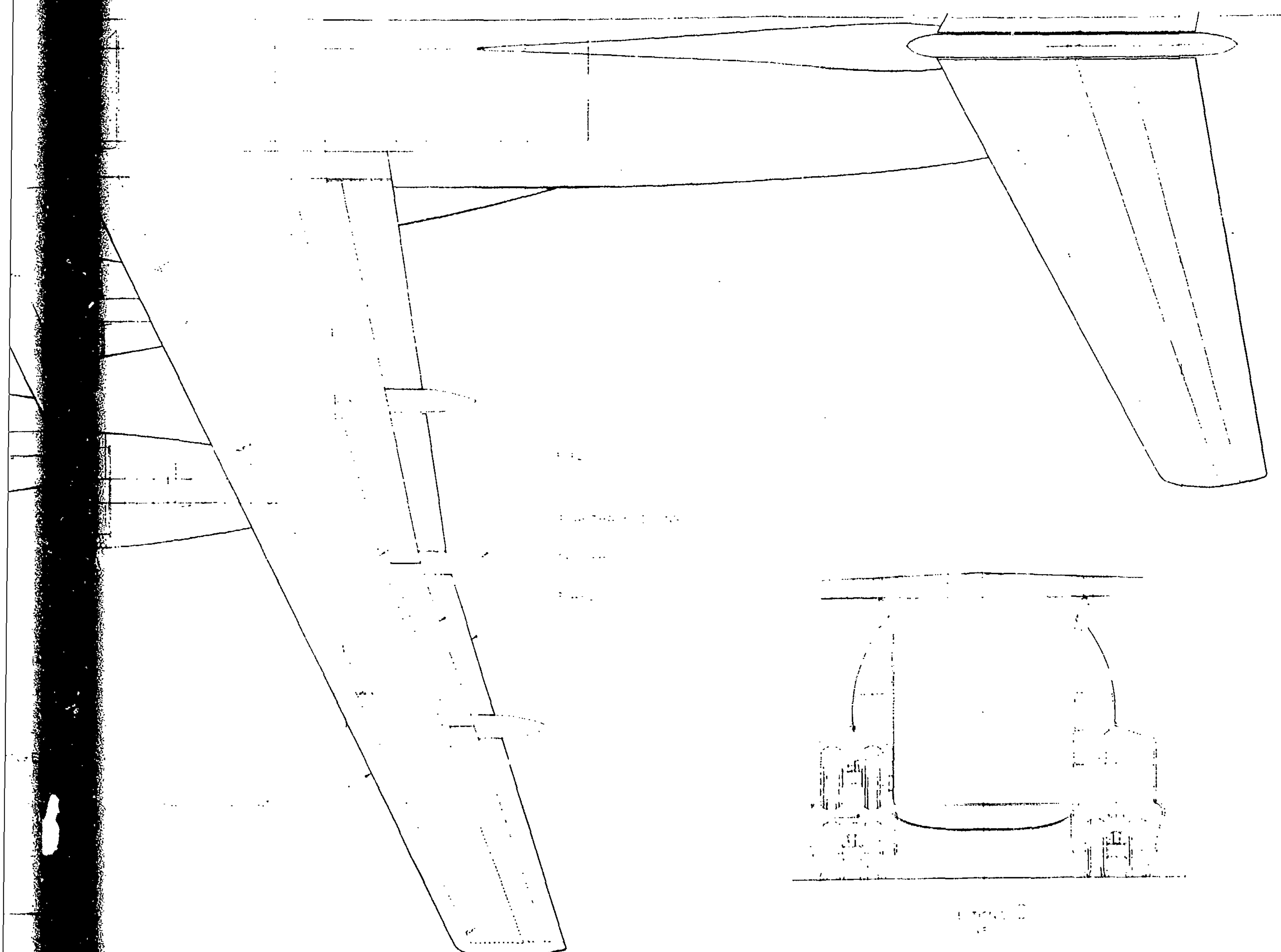

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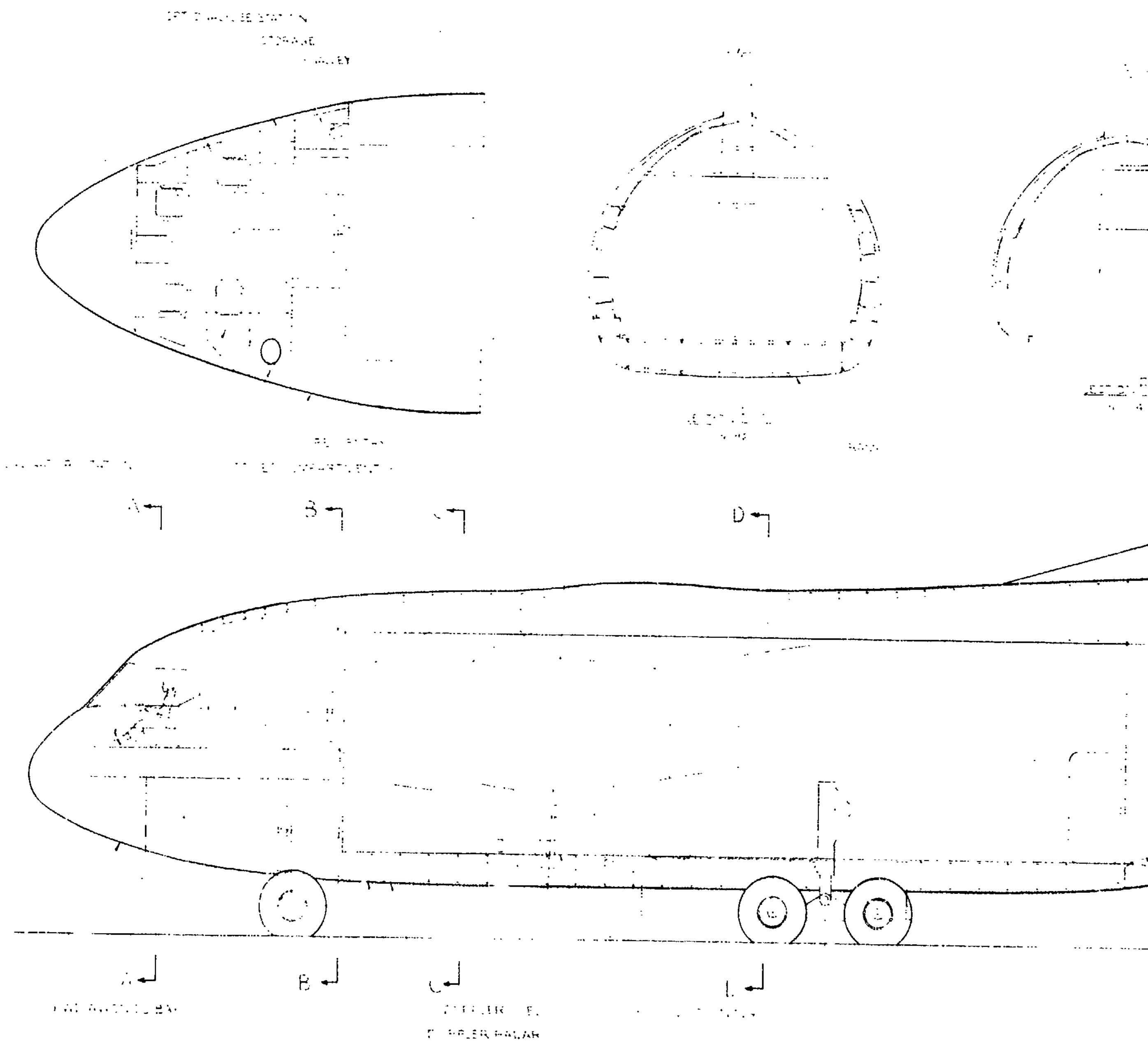

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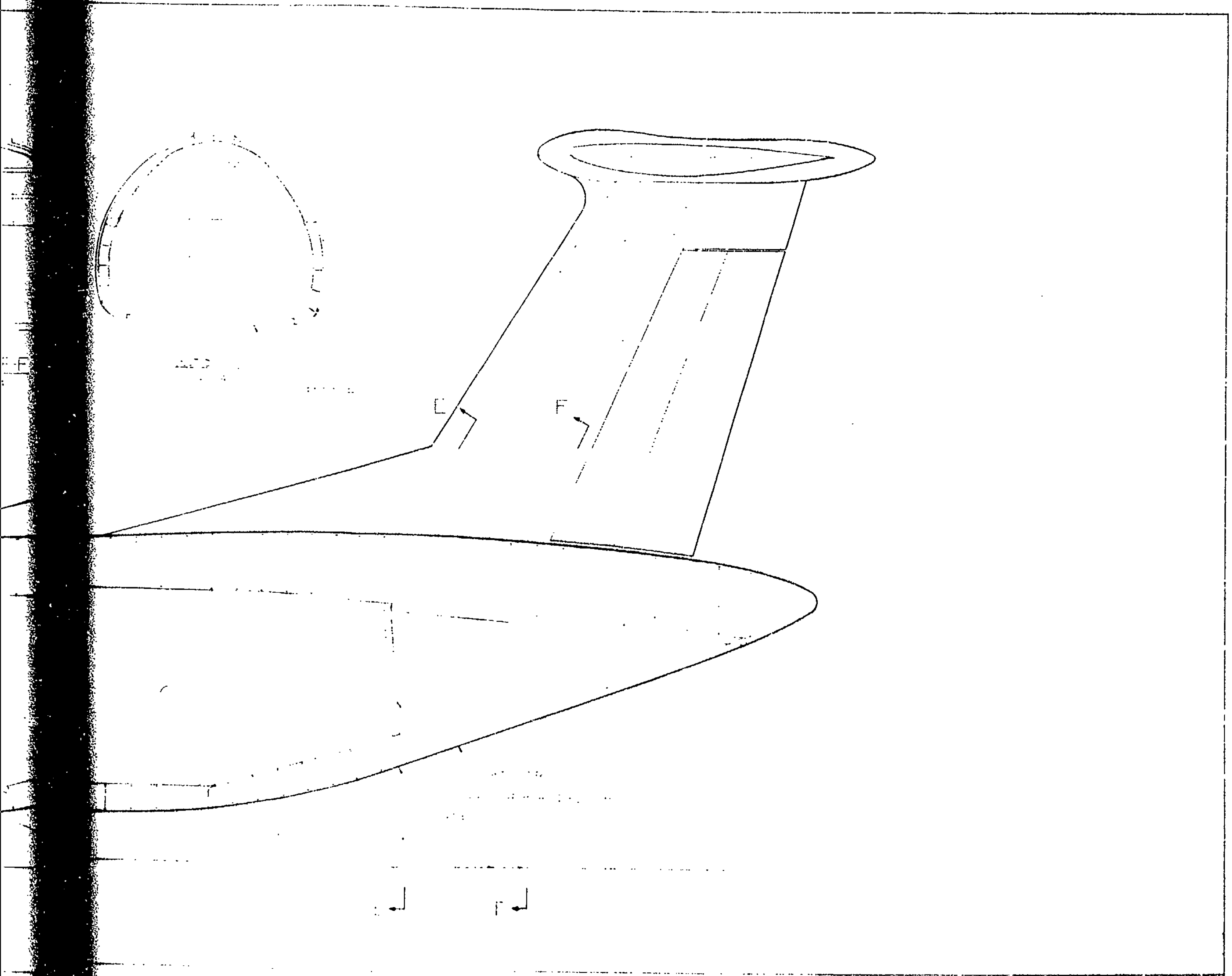

Figure 25. Refined Basepoint Configuration Inboard Profile

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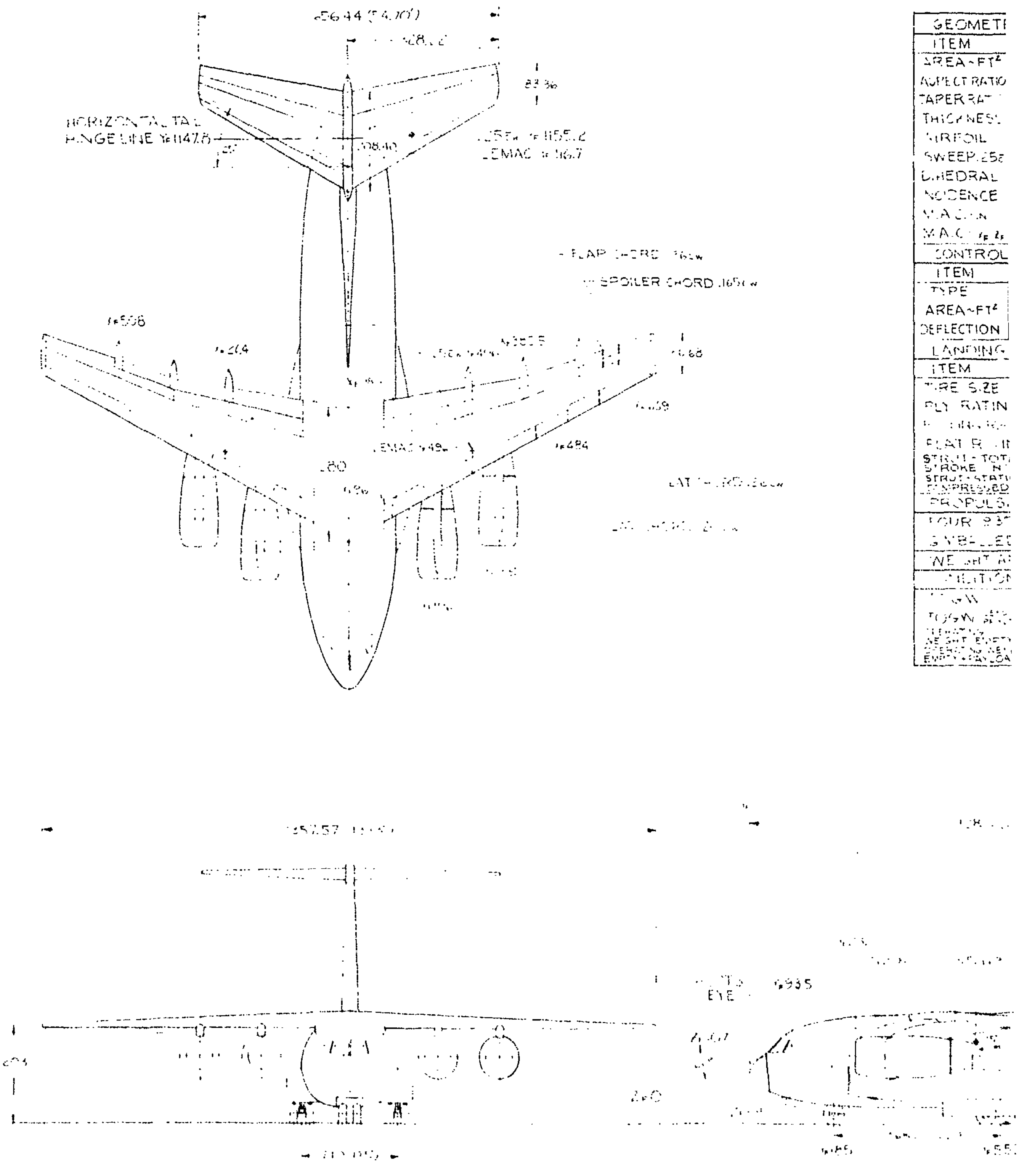

HORIZONTAL TAIL
HINGE LINE Y=1147.8
LEMAC
FLAP CHORD
SPOILER CHORD
GEOMETR
ITEM
AREA~FT²
ASPECT RATIO
TAPER RAT
THICKNESS
AIRFOIL
SWEEP .25c
DIHEDRAL
INCIDENCE
CONTROL
ITEM
TYPE
AREA~FT²
DEFLECTION
LANDING
ITEM
TIRE SIZE
PLY RATIN
PROPULS
EYE

Figure

GEOMETRIC DATA				
ITEM	WING TRAP	WING TOTAL	HORIZONTAL	VERTICAL
AREA - FT²	1430	1600	665	410
ASPECT RATIO	8.95	8	4.50	1.40
TAPER RATIO	0.40		0.40	0.60
THICKNESS	.125 [illegible] TIP		12 - 08	12 - 10
AIRFOIL	64A[illegible] - 64A310		64A212 CR 64A208 CT	64A212 CR 64A210 CT
SWEEP [illegible]	25°		25°	30°
DIHEDRAL	0° LOWER [illegible]		0°	~
INCIDENCE	+4° [illegible] +1° CT		VARIABLE	~
M.A.C. [illegible]	161	159	54.8	209.64
M.A.C. [illegible]	291	274	40.67	131.77

CONTROL SURFACE DATA					
ITEM	FLAP	SPOILER	SLAT	RUDDER	HORIZ
TYPE	DOUBLE SLOTTED	UPPER SURFACE ONLY	POWERED	DOUBLE HINGED	ALL MOVABLE [illegible] DOUBLE-HINGED
AREA - FT²	[illegible]	193.5	205.6	148.6	625
DEFLECTION	[illegible] 80° [illegible] 50°	60°	[illegible] 38° [illegible]	[illegible] SEGMENT	[illegible] SEGMENT

LANDING GEAR DATA		
ITEM	MAIN	NOSE
TIRE SIZE	475 × 725 20 [illegible] TANDEM	475 × 725 20 [illegible] TWIN
PLY RATING	12	12
[illegible]	[illegible]	[illegible]5
[illegible]	24	12.4
STRUT TOTAL STROKE [illegible]	3[illegible]	3[illegible]
STRUT STATIC [illegible]	5	5

PROPULSION DATA
FOUR [illegible] SIZE GENERAL ELECTRIC 3[illegible] ENGINES
[illegible] NOZZLES WITH [illegible] DEFLECTION

WEIGHT AND [illegible] DATA					
[illegible]	WEIGHT	[illegible]	Z [illegible]	% MAC [illegible]	% MAC [illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]35
[illegible]	[illegible]	504.[illegible]	27.2	[illegible]6.77	[illegible]67
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	54.78
[illegible]	126149	[illegible]	[illegible]	29.73	[illegible]

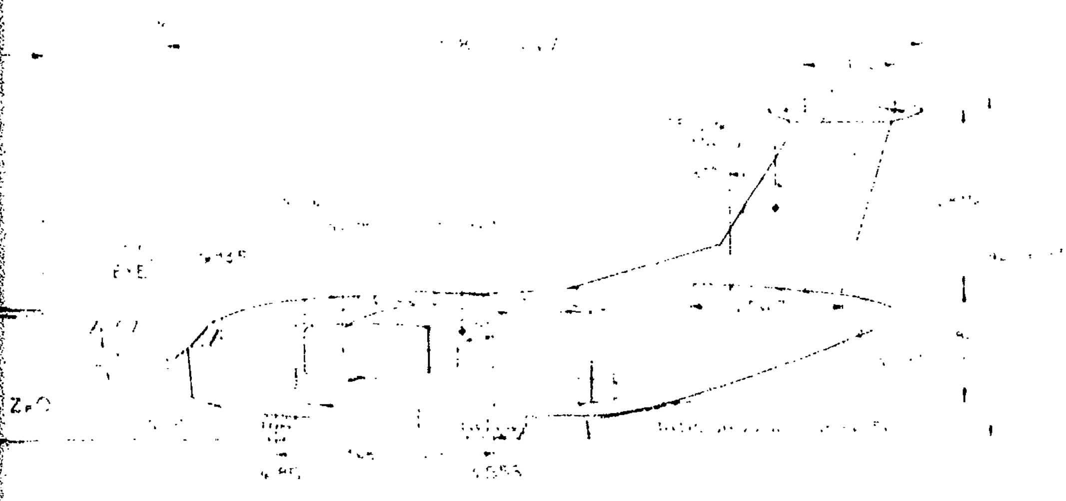


Figure 26. Refined Basepoint Configuration - 3 View

the wing chord plane. The chord of the outboard one-third of the slat is 26 percent of the wing chord; this section can be extended 38 degrees below the wing chord plane.

The double-slotted flaps shown are 36 percent of the wing chord and are consistent with those used on the wind tunnel model. The results of the vehicle sizing analysis, using current wind tunnel data, showed that the takeoff and landing criteria were not critical. Flap chord length and area could be reduced since the takeoff and landing sizing conditions do not dominate the wing design through wing loading. Second-order weight reductions would be expected if another design iteration were performed. The forward flap segment is 16 percent of the wing chord with 30 percent aft segment. Forward flap maximum extension is 25 degrees with the aft flap extending to 80 degrees. The flap system is divided into three spanwise segments over approximately two-thirds of the exposed span. The aft flap can be fast actuated upward 60 degrees from the fully extended position to provide a waveoff capability during the final landing approach. Wing spoilers provide the necessary lift control during the landing maneuver.

The aileron is a double-panel flaperon system, with the total system chord equal to 36 percent of the wing chord. The aft panel is 30 percent of the wing chord, with a conventional 15 percent chord plain-hinge aileron segment used during cruise flight and climb, or descent with flaps retracted. When the flaps are extended, the forward flaperon panel is deflected 25 degrees to a fixed position and the aft panel is deflected to a neutral position of 50 degrees. From this position, the insert aileron can be deflected ± 25 degrees for roll control, in combination with spoilers.

Vehicle maximum fuel capacity is 63,992 pounds, with 60,100 pounds fuel in integral wing tanks and 3,892 pounds in forward wing center section fairing bladder tanks. The 500-nautical-mile radius design mission fuel required is 33,161 pounds. Maximum fuel capacity is determined by the 2,600-nautical-mile deployment mission. The drawing as shown uses self-sealing bladder tanks ahead of the wing front spar in the wing-fuselage fairing. Installing fuel in this manner is the most efficient method of obtaining the fuel capacity required. To completely install all the fuel between the front and rear spars would require a 78-square-foot wing area increase. After resizing, takeoff gross weight would increase by approximately 2,150 pounds. The bladder tank and its associated installation effects resulted in a 400-pound increment in gross weight.

The refined configuration inboard engine centerline has been located one nacelle maximum diameter from the fuselage, and the two engines centerlines are separated by 1-1/2 diameters. The nozzle is placed at the wing leading edge to maximize blowing efficiency on the flaps for STOL operation. A gimbaled nozzle control the mixed hot and cold exhaust air through a range of ± 15 degrees. During the takeoff ground roll, the exhaust is deflected 15 degrees

down, then changed to 15 degrees up at lift-off and during climb until the flaps are retracted. This is to maximize acceleration during the takeoff and ground roll, and maximize lift at lift-off. The exhaust is also deflected 15 degrees during landing approach when the flaps are extended. With flaps up during climb, cruise, or descent, the exhaust is undeflected. Cascade-type thrust reversers are provided for the fan air only. Turbine thrust is reduced during fan thrust reverse due to the apparent increase in nozzle area resulting from elimination of the fan exhaust in this mixed-flow nozzle.

The main landing gear is a dual-tandem design which retracts into pods off the lower shoulder of the fuselage in a manner conventional for high-wing transport aircraft. The tires have been sized for a CBR 6 and 200 passes at midmission landing weight and design payload of 28,000 pounds. Design sink speed is 15 feet per second at midmission weight. Reduced sink speed is used at higher air vehicle weight. The nose gear is a two-wheel dual, forward retracting design. Although analysis shows that a smaller tire could be used on the nose gear, tires have been used identical to the main gear to simplify logistics and improve airplane ground handling. The nose gear is steerable to permit turning the airplane on a 60-foot-wide strip.

The empennage is a T-tail arrangement, with the horizontal surface placed slightly higher and further forward than conventional T-tails. Both the vertical and horizontal tails have double-hinged control surfaces with hinges at 55- and 75-percent chord. Each panel deflects 25 degrees independently, making a total of 50 degrees relative to the centerline for the aft panel when both panels are fully deflected. Both panels are used for control during STOL operations; however, for conventional flight, the forward panel is locked and the aft panel only is used for control. The fixed horizontal surface can also be adjusted +10 and -20 degrees for airplane longitudinal trim. The horizontal tail is an inverted 64A200 section with a 15-percent leading edge device that can be extended upward 15 degrees for additional pitch control during one-engine-out waveoff maneuver. Horizontal tail thickness ratio at the root is 12 percent with an 8-percent thick trip. The vertical tail uses a 64A000 airfoil section and is 12 percent thick at the root. The tip chord is 10 percent thick.

4.2 EXTERNAL GEOMETRY

4.2.1 Wing and Empennage Surfaces

	Wing	Horizontal Tail	Vertical Tail
Area (sq ft)	1,600	665	410
Span (in.)	1,357.6	656.4	287.5
Taper ratio	0.40 (trapezoid)	0.40	0.60
t/c root	12.5	12	12
t/c tip	10.0	8	10
Camber	0.0237	0.0158	0
Airfoil Section	64A3	64A2	64A0
Spanwise twist	+4°+1°	-	-
MAC (in.)	189	154.8	209.6
MAC location	X_F 274	X_F 140.7	Z_F 131.8
Sweep c/4	25°	25°	30°
Root chord (in.)	280 (total) 216.7 (trapezoid)	208.4	256.7
Leading edge angle	27.25°	29.1°	33.5°
Trailing edge angle	9° to X_F 387, 18° op	10.0°	17.0°
Wetted area (sq ft)	2,634 ob of X_F 77	1,262	918
Tail arm (in.)	-	659	530
Incidence	+4° root, +1° tip	Variable	-
Dihedral	0° lower surface	0°	-

4.2.2 Fuselage

Overall length (ft)	103.08
Cross-section area (sq ft)	226
Constant area section length (ft)	45
Forebody fineness ratio	1.44
Aft fuselage fineness ratio	2.34
Cross-section shape variation	Circular
Pressurized volume (cu ft)	16,046
Operating differential pressure - psi	8.8
Wetted area (sq ft)	4,208

4.2.3 Nacelle

Wetted area (sq ft)	320
Number	4
Capture area to max cross-section area ratio	0.635
Length (in.)	195
Maximum diameter (in.)	88
Exit area to max cross-section area ratio	0.41
Ratio of distance from inlet to max cross-section/total length	0.323
Nacelle pylon vertical length (in.)	22
Nacelle pylon width (in.)	47.0
Nacelle pylon sweep (c/4)	35°
Nacelle pylon wetted area (sq ft)	73.8

4.2.4 Landing Gear Housing

Wetted area (sq ft)(per side)	436.5
Length (in.)	585.0
Maximum equivalent diameter (in.)	63.5

4.2.5 Flap Geometry

Flap chord/wing chord ratio	36%
Flap span (in.)	581
Spanwise location - inboard end X_F	99
Type of flap	Double-slotted
Fowler motion	14%

4.2.6 Spoiler Geometry

Spoiler chord/wing chord ratio	16.5%
Spoiler span/wing span ratio	85.7%
Wing chordwise location - LE	67.5%
Spanwise location - inboard end X_F	99

4.2.7 Propulsion

Engine type	GE turbofan
Engine designation	93 percent GE 13/F10
Number of engines	4
Sea level static thrust	
Installed (lb)	20,540
Uninstalled (lb)	22,320

Max static rated thrust - 2,500 ft, 93.4° F	
Installed (lb)	19,800
Uninstalled (lb)	21,530
Bypass ratio	6.5
Engine weight (lb)	2,740
Engine diameter (in.)	71.0
Engine length (in.)	90.3
Mechanical power extraction (hp)	80
Air-bleed requirements (lb/sec)	Compressor: 1 Interstage: 15

4.2.8 Landing Gear

Layout of landing gear on aircraft	Refer to paragraph 10.6
Load distribution	
Percent on nose gear	22
Percent on main gear	78
Main gear	
Extended length, top of strut to axle (in.)	107
Stroke (in.)	30
Tire designation and size	47.50 x 17.25 - 20 12 PR
Number of tires	4
Tire arrangement	Twin - tandem
Distance apart fore and aft (in.)	73.2
Distance apart sideways (in.)	33.0

Nose gear	
Extended length, top of strut to axle (in.)	80
Stroke (in.)	25
Tire designation and size	47.50 x 17.25 - 20 12 PR
Number of tires	2
Tire arrangement	Twin
Distance apart fore and aft (in.)	-
Distance apart sideways (in.)	33
Maximum aircraft gross weight - deployment	191,172 pounds
Design gross weight - employment	159,310 pounds
Midmission weight - (CBR 6, 200 Passes)	145,238 pounds
Antiskid system allowable slip for operation on:	
Hard surfaces	25%
Soil	35%
Maximum allowable nose gear angle on:	
Hard surfaces	77° 15'
Soil	77° 15'
Maximum practical steer angle to be used on soil runway	Unknown
Max breaking energy available from wheel brake system (ft-lb)	24,400,000
Max braking energy available per main wheel (ft-lb)	3,050,000

4.2.9 Miscellaneous

Cargo floor design loading	
Entire floor minimum (lb/sq ft)	300 limit
Under wing floor (lb/sq ft)	400 limit
Door dimensions	
Front crew entry (in.)	30 x 60
Aft crew entry (in.)	37 x 72
Escape hatches (in.)(7 total)	22 x 22
Aft loading ramp (in.)	140 x 200
Aft cargo door (in.)	140 x 220
Static cargo floor height above ground (in.)	56
Pilot vision angle over nose	22°

SECTION V

PERFORMANCE AND SIZING

5.1 MISSIONS

The mission calculations conform to the mission descriptions as presented in paragraph 2.3 Fuel allowances for warmup, taxi, conventional takeoff, and reserve are in accordance with MIL-C-5011A. For the STOL takeoff at the midpoint of the employment mission, fuel allowance for warmup and takeoff is the fuel used in 5 minutes with normal rated power at sea level, std. day. A 5-percent service tolerance on fuel flows is applied, and no credit is taken for time, fuel, or distance during descents.

Missions and mission trades data are calculated using NR-developed computer program FK800. The U.S. Standard Atmosphere, which is built into the program along with other atmospheres, is used for mission determination. The program is used also for generation of independent printouts of climb, cruise, loiter, and other mission element data. Input data for the computer programs include the installed engine performance, total aircraft drag variations with lift coefficient and mach number, and weight information. The data are obtained from paragraph 6.1.1.

A plot of payload-versus-radius for various gross weights is presented in figure 27 for employment mission details. A takeoff gross weight for a 3.0 g load factor is shown, along with reduced takeoff weights of 10,000-pound decrements. The basic mission radius of 500 nautical miles with a 28,000-pound payload is indicated on the plot. A similar plot for a takeoff gross weight for a 2.5 g load factor and takeoff gross weights reduced by 20,000-pound decrements is presented in figure 28.

Similarly, a plot of payload-versus-range for various gross weights is given in figure 29 for deployment mission details where the basic takeoff weight is limited by a 2.5 g load factor. The basic mission requiring a 2,600-nautical-mile range and the resulting payload is indicated on the plot. A similar plot for a takeoff gross weight for a 3.0 g load factor is presented in figure 30. A ferry mission (deployment mission with 2.5 g takeoff weight without cargo) equal to 4,222 nautical miles is obtained which exceeds the required 3,600 nautical miles. This ferry range is obtained by adding bay tank fuel in the cargo compartment.

A plot of range prediction along with the corresponding cruise altitude is presented in figure 31. The data are for the cruise mach number of 0.75. Figures 32 through 35 present thrust required and specific range as a function of mach number and gross weight. The plots are for level flight cruise at altitudes of 10,000, 20,000, 30,000 and 40,000 feet.

Tabulated data of the calculated basic employment, deployment, and ferry mission details are presented in tables XI, XII, and XIII.

5.2 FLIGHT ENVELOPE

Flight envelope data at given gross weights are determined by use of the NR developed flight envelope program 2A700. Figure 36 presents speed altitude envelopes using intermediate power for the minimum flying gross weight, midpoint gross weight of the employment mission, employment mission takeoff weight, and deployment mission takeoff weight (maximum gross weight). The minimum flying gross weight is the weight with a reserve fuel and payload at the end of the employment mission. It is noted that the maximum speed is the aircraft q limit up to 23,500 feet. Above this altitude, the aircraft is thrust-limited at near the maximum mach number limit.

5.3 STOL TAKEOFF AND LANDING

The STOL takeoff and landing distances are presented in figure 37 for hot day conditions at 2,500 feet altitude. The distances are based on the STOL takeoff and landing criteria listed in paragraph 3.2, i.e., one-engine-out safety margins are included as well as waveoff capability in case of engine failure. The takeoff distance is computed without obstacle constraints, and the landing distance includes a 50-foot height over the threshold (not a 50-foot obstacle) and is based on a 10-feet-per-second sink rate above the threshold without flare.

The distances shown represent the travel of the point mass of the aircraft, i.e., no fuselage length is added, no taxi-on or taxi-off distances are included, and no factor for an increase of the field length is used.

At the high gross weight, the takeoff distance is a balanced field distance. At lower gross weights, the takeoff distance is determined on the basis of an accelerate-and-stop distance where the failure speed V_F is based on the ground control speed V_{mc_g} rather than on the balanced field concept.

The acceleration phase is based on a friction coefficient of $\mu = 0.10$, and an engine nozzle deflection of 15 degrees downward in such a way that the exhaust is not obstructed by a flap. This deflection yields a high forward acceleration. The deceleration phases are based on a friction coefficient of $\mu = 0.30$ and thrust reversal of two engines, assuming that one engine has failed and the opposite engine is idled to preserve thrust symmetry. A reversal efficiency of $\eta = 50$ percent is taken in accordance with the ground rules stated in paragraph 3.2.

The ground control speed, failure speed for balanced field length, liftoff speed, and approach speed are presented in figure 38. The liftoff speed meets the climb requirement of 3 degrees with the critical engine failed. The approach speed, V_a, is lower than the liftoff speed; however, this does not negate the waveoff capability. During waveoff, the second segment of the double-slotted flap will be retracted partially, and this generates an adequate forward acceleration so that the speed will be at least equal to the liftoff speed above the far end of the runway as is shown in volume III. A retraction rate of 12 degrees per second is provided. During this retraction, the forward flap segment will not move and will be kept at a 25-degree angle. Adequate ground clearance is available to demonstrate a speed equal to $V_{mtd} = V_a/1.1$ in ground effect as a minimum for the approach with one engine failed.

The takeoff and landing distances and the liftoff and approach speeds are determined in accordance with the aforementioned volume III. The aerodynamic data basis is described in paragraph 6.1.1.

5.4 SIZING

The development of the basic configuration described in section IV was accomplished by using the design criteria of section III in conjunction with existing NR computer programs. Vehicle sizing involving the selection of a specific thrust-to-weight ratio and wing loading combination for a specific mission radius to obtain the desired takeoff and landing performance were done using the vehicle sizing and performance program (VSPEP). This program computes performance characteristics such as mission radius or range on several missions. It also is capable of scaling the computer basepoint for defining related vehicles having different wing areas (or wing loadings), engine size (or thrust-to-weight ratios), gross weight, payload weights, and payload sizes. This program has the capability to determine vehicle gross weight which satisfies a specified radius or range mission requirement at desired engine sizes and wing areas.

The configuration sizing process was started with a modified version of the preliminary baseline configuration. This vehicle was updated to include changes in study ground rules and the results of the various studies that have been conducted during the part 2 portion of the program. The vehicle sizing program basepoint included an adjusted weight estimate, minor geometric changes, and the revised aerodynamic characteristic representative of the update preliminary baseline. Sizing of the vehicle for the various design conditions was done for the thrust-to-weight and wing loading combinations that satisfies the predominate design requirement. The results of sizing are vehicle takeoff gross weight mission fuel requirements, engine size, and vehicle wing areas. These design parameters are used in the configuration layout process which precedes the configuration as shown in section IV. The

dominant configuration design requirements are the wing fuel volume determined by the 2,600-nautical-mile range mission and the speed requirement at mach 0.75 at 20,000 feet altitude.

Figure 39 depicts the various sizing criteria on a thrust-to-weight versus wing loading plot. The previously mentioned fuel requirement, mach 0.75 speed requirement, and the takeoff and landing requirements are shown. Values of thrust-to-weight and wing loading to the left of the boundary of fuel available equal to fuel required, and above the mach 0.75 at 20,000-foot boundary, meet or exceed the desired performance. The lightest aircraft that meets the STOL-TAI design requirements is at the thrust-to-weight and wing loading determined by the intersection of these two design criteria. The vehicle resulting from the thrust to weight of 0.516 and wing loading of 99.3 pounds per square foot is described in section IV. This vehicle weighs approximately 159,310 pounds. A vehicle sized to meet the takeoff requirement and the fuel available/required would be about 9,000 pounds lighter, but would not be capable of flying mach 0.75 at 20,000-foot altitude. The optimum cruise speed at altitude would reduce to mach 0.685 from mach 0.75. Landing distance over a 50-foot obstacle would be approximately 1,725 feet. No further relaxation in design criteria would appear to be feasible for a STOL aircraft since the next most important design criteria is the landing distance. A reduction of thrust-to-weight ratio would reduce takeoff gross weight, but the takeoff ground rules would be violated; therefore, a lower value would not be practical without further operational requirements considerations.

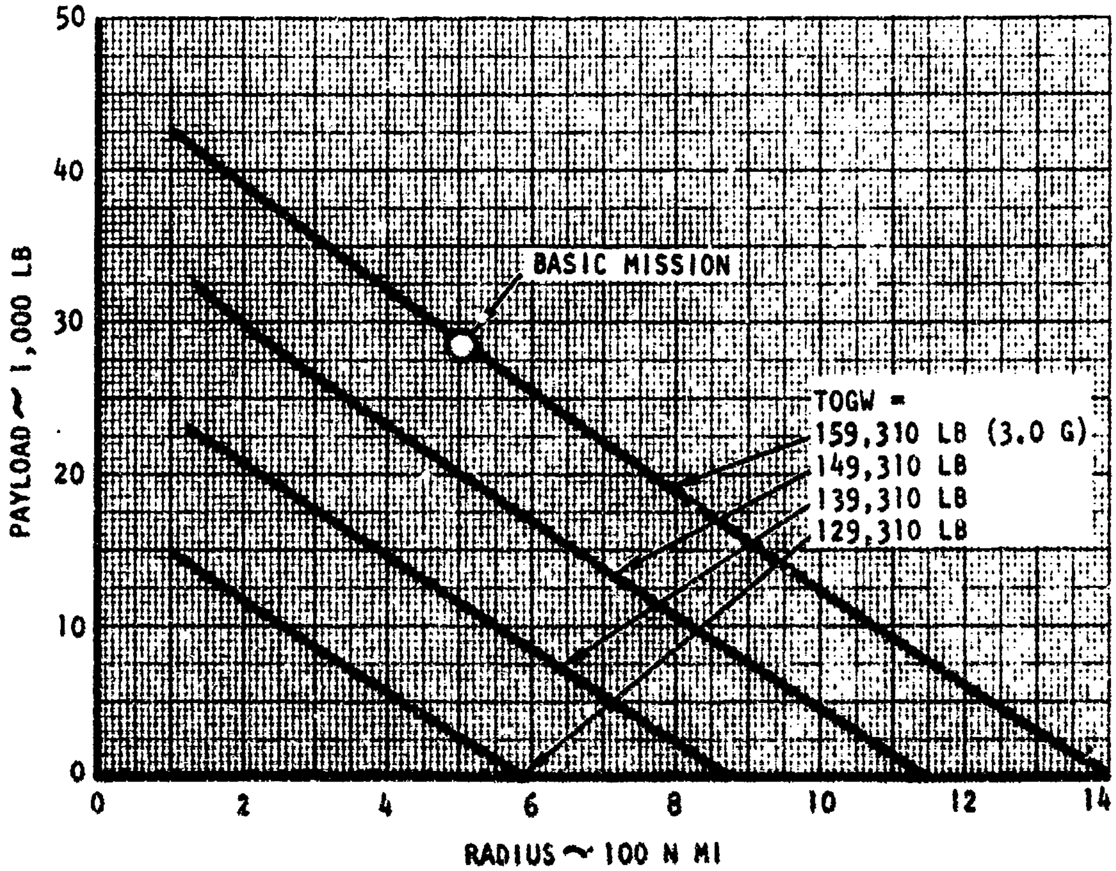


Figure 27. Employment Mission Payload-Radius Trade

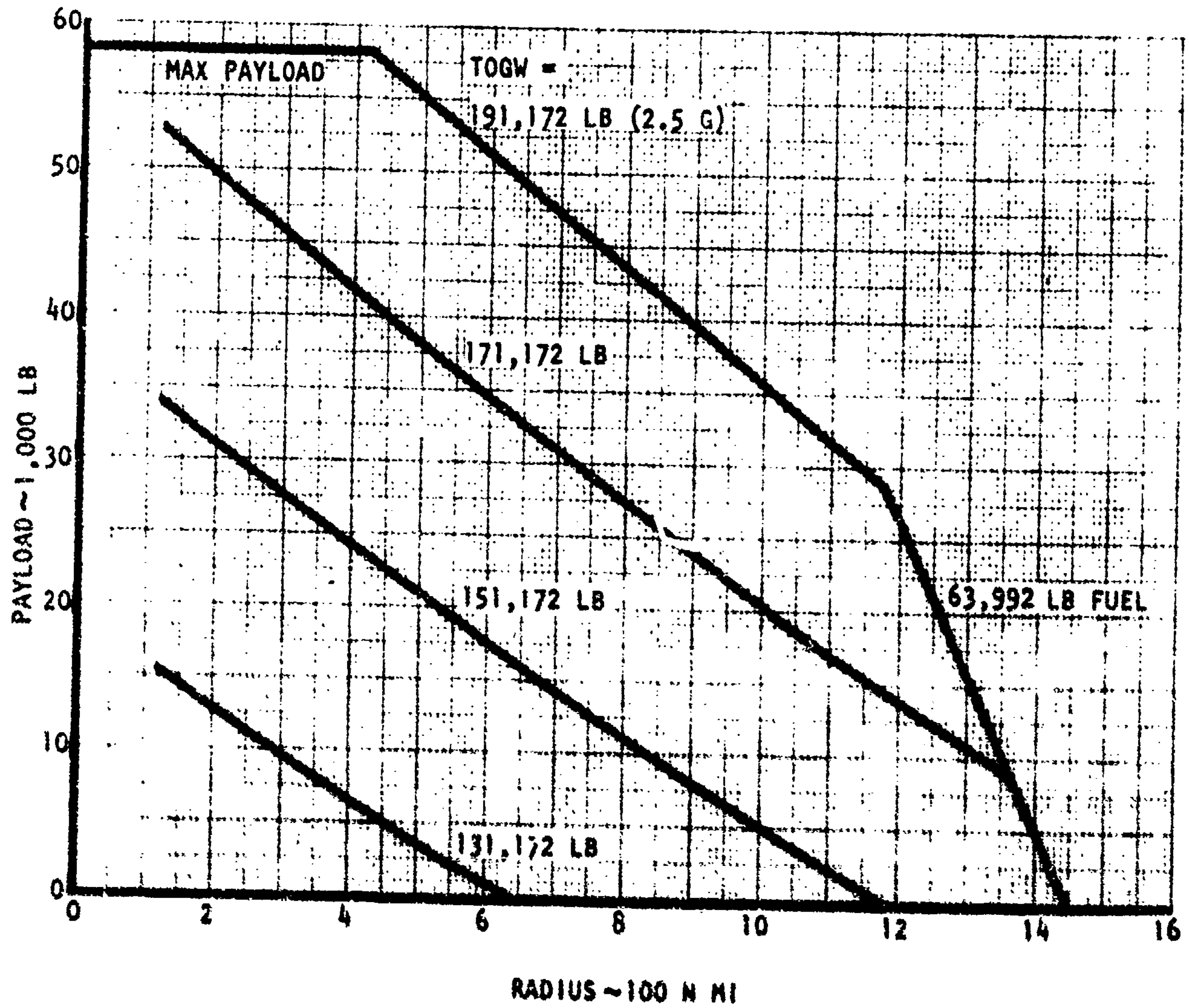


Figure 28. Employment Mission Payload-Radius Trade

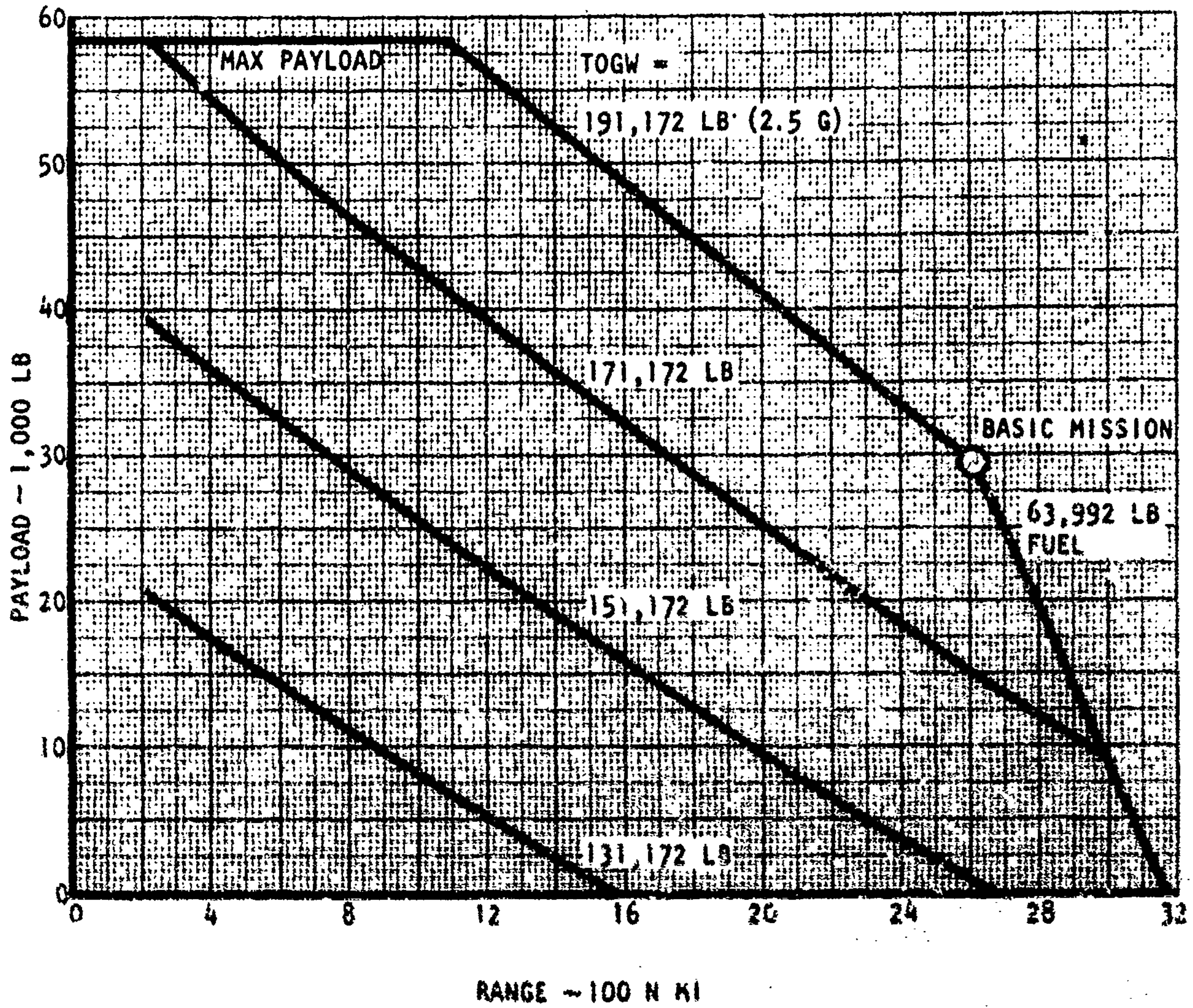


Figure 29. Deployment Mission Payload-Range Trade

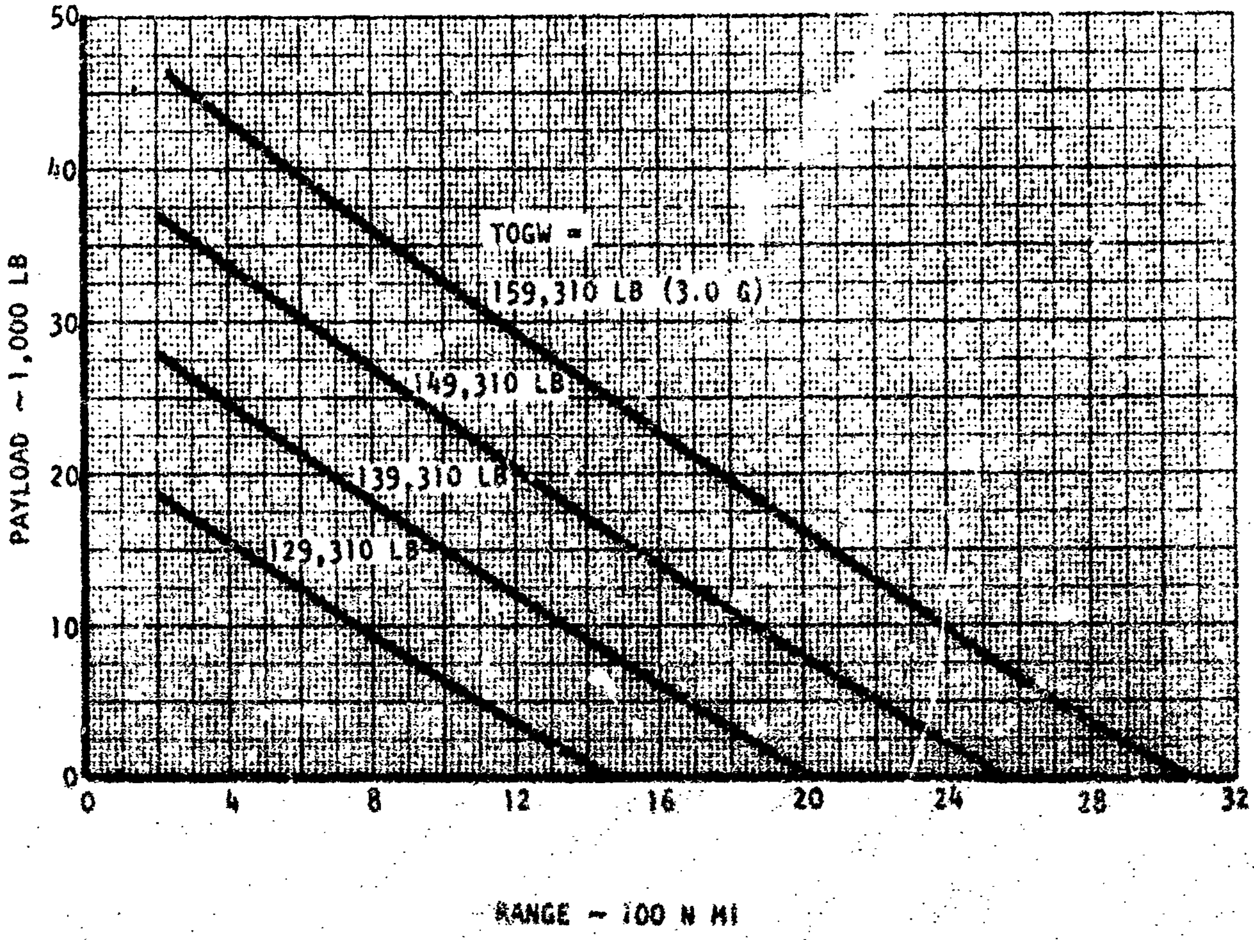
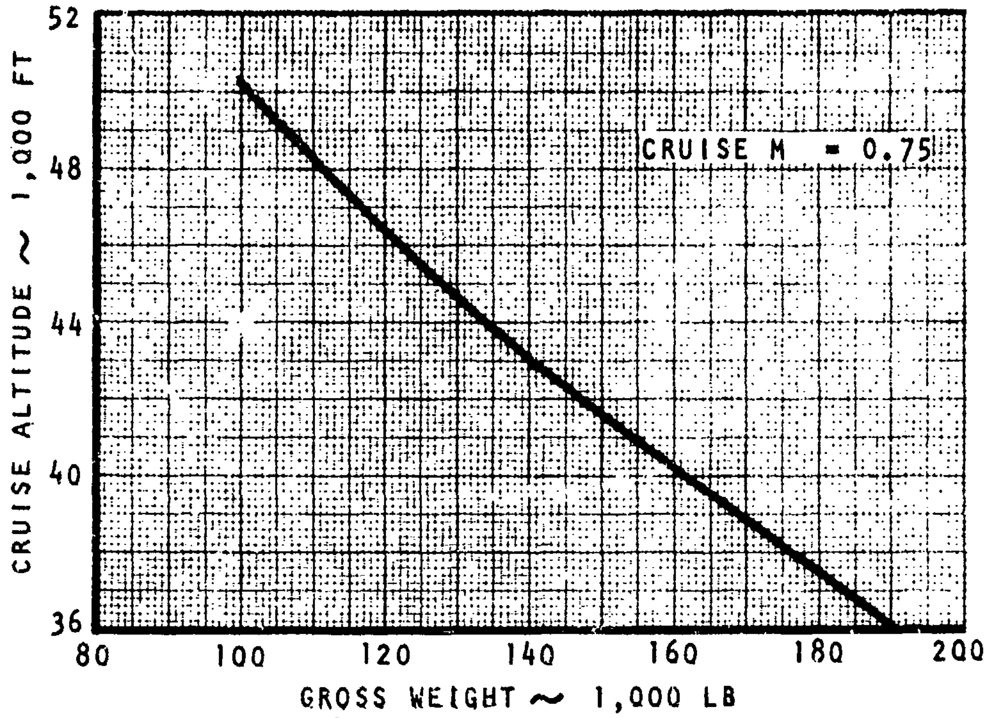


Figure 30. Deployment Mission Payload-Range Trade



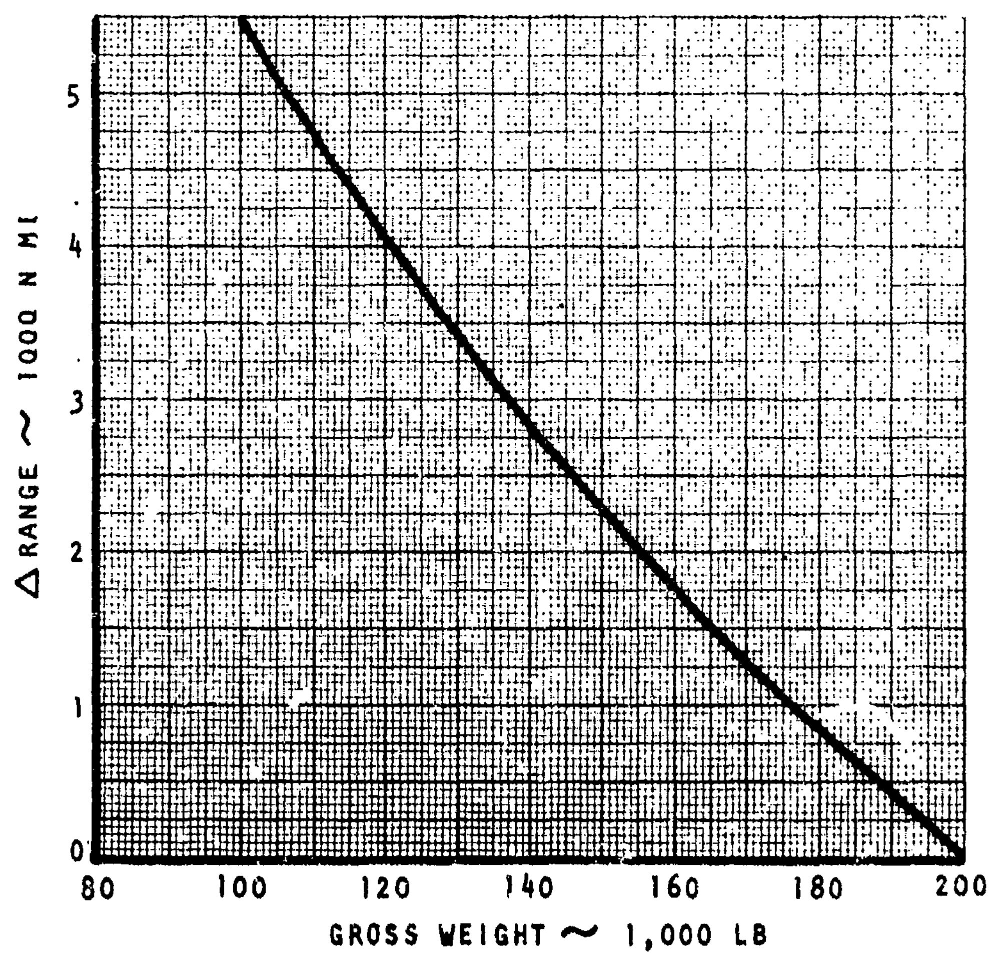


Figure 31. Range Prediction

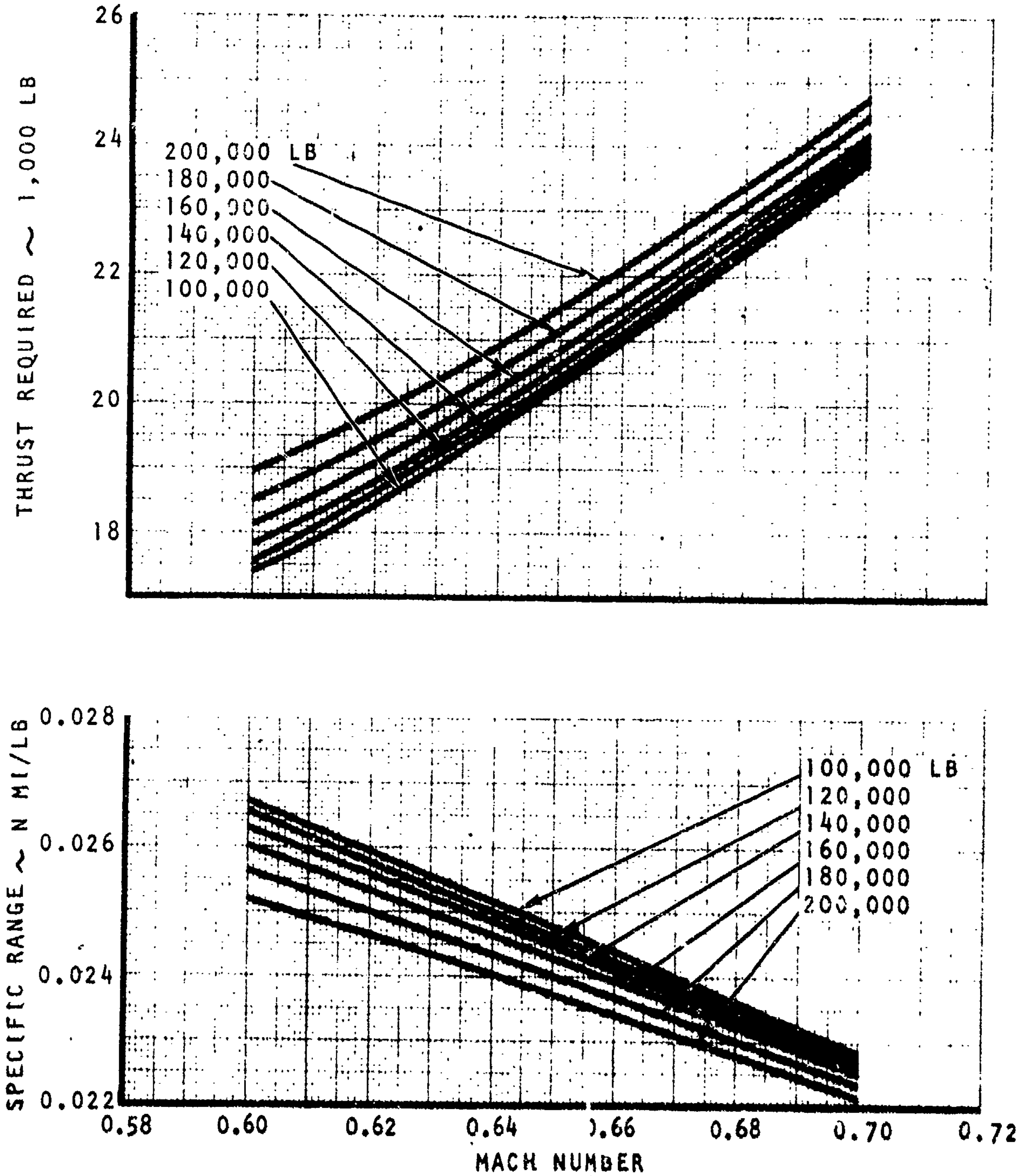


Figure 32. Thrust Required and Specific Range Cruise at 10,000 Feet

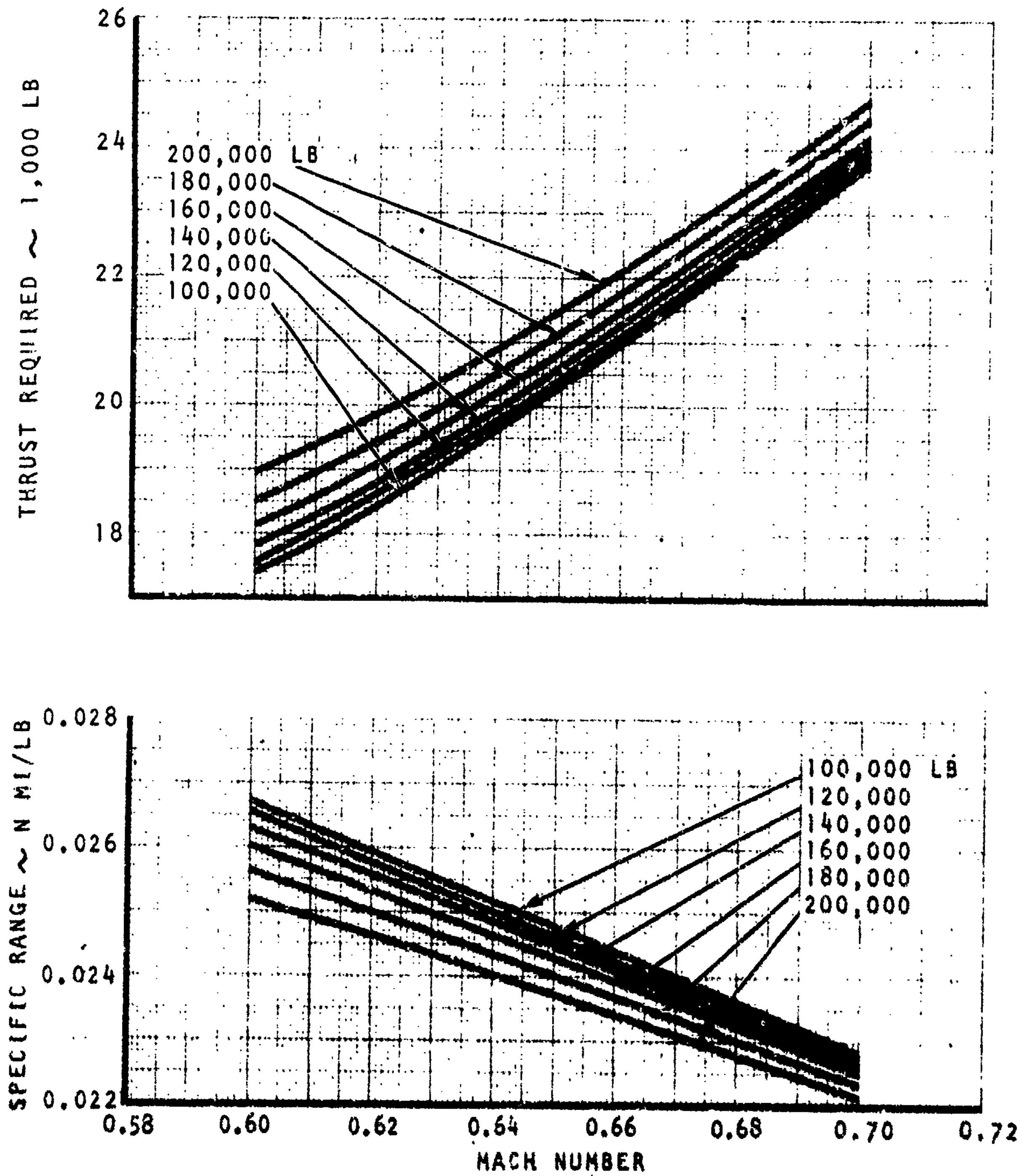


Figure 32. Thrust Required and Specific Range Cruise at 10,000 Feet

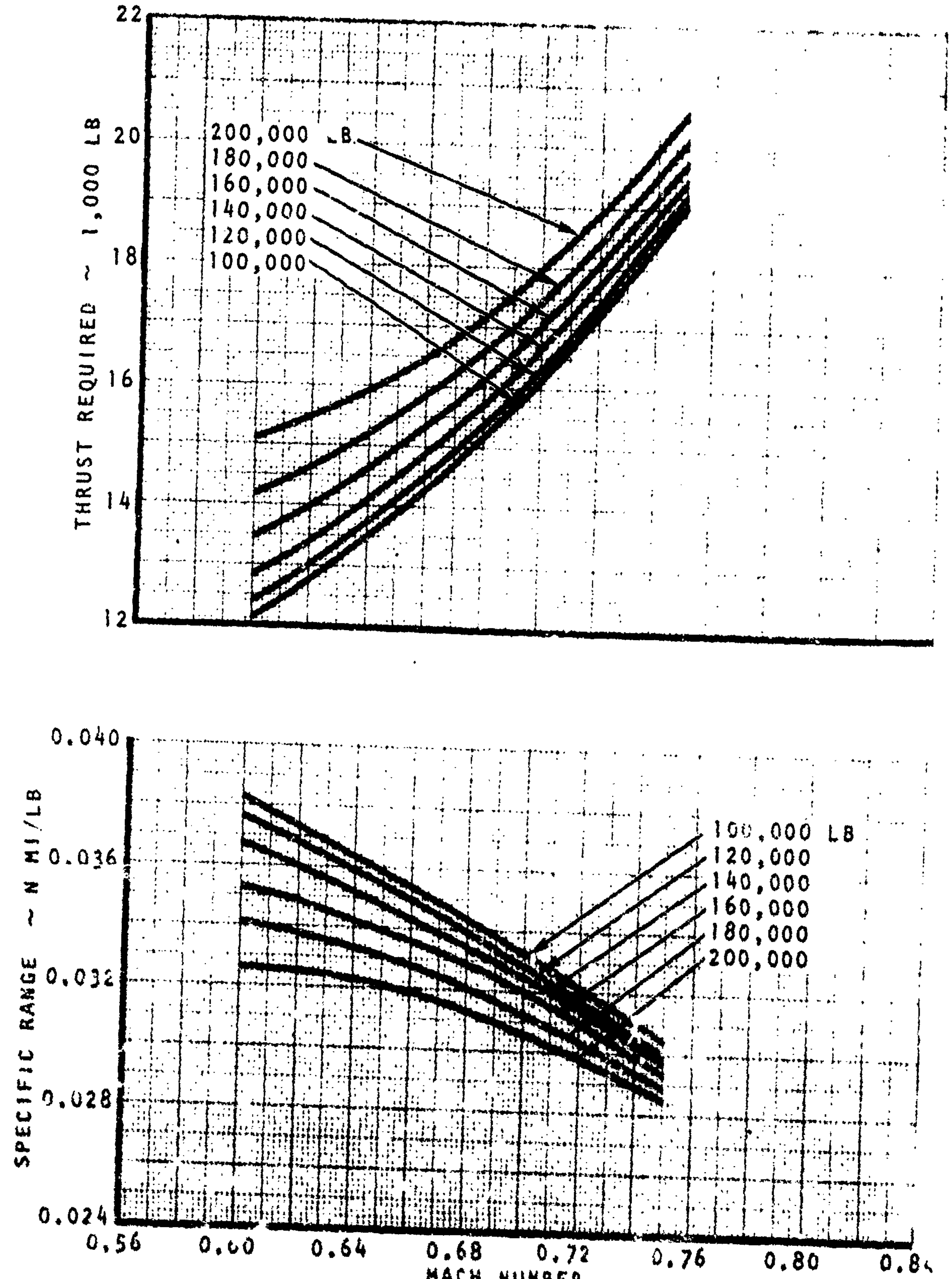


Figure 33. Thrust Required and Specific Range Cruise at 20,000 Feet

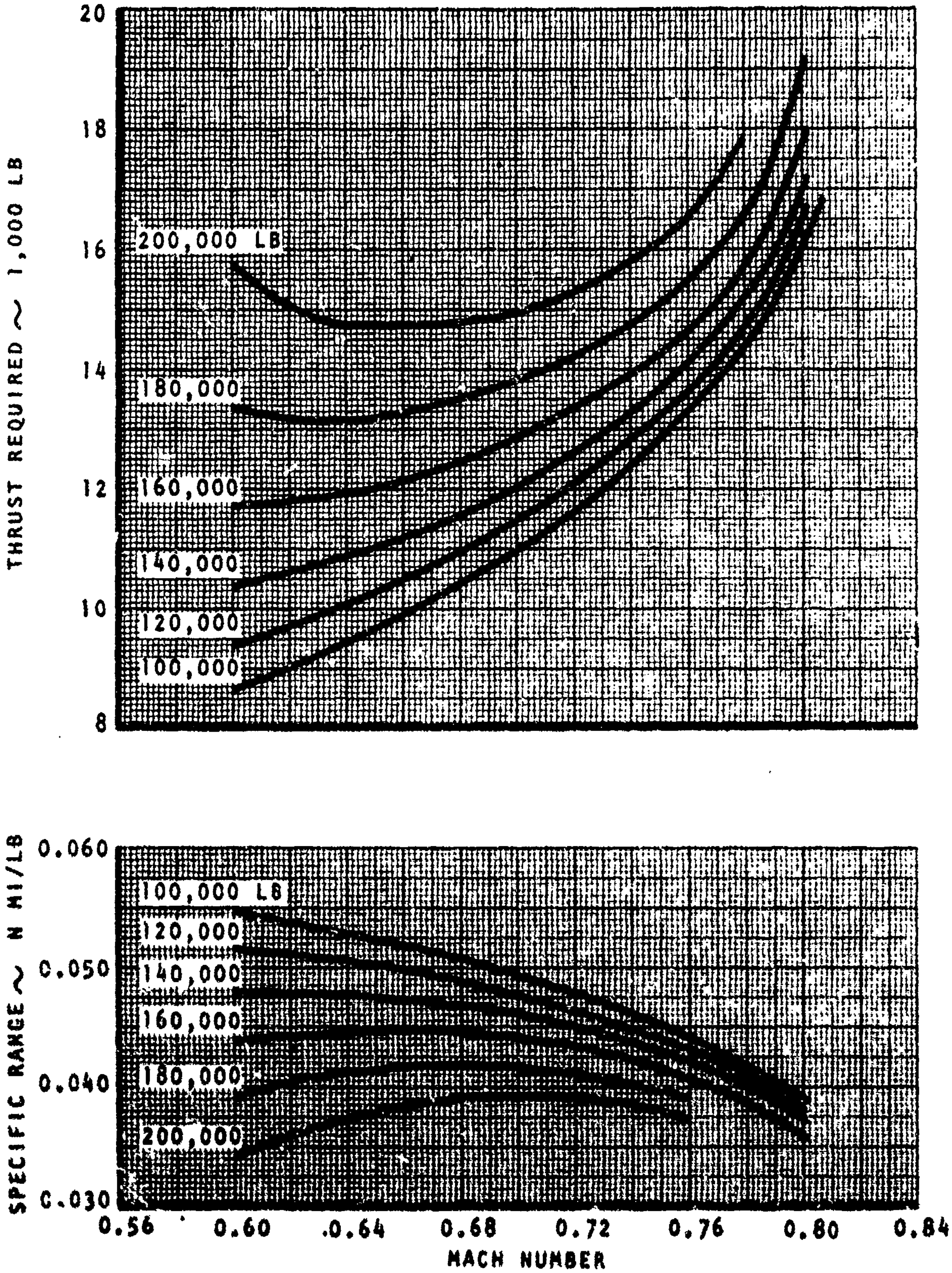


Figure 34. Thrust Required and Specific Range Cruise at 30,000 Feet

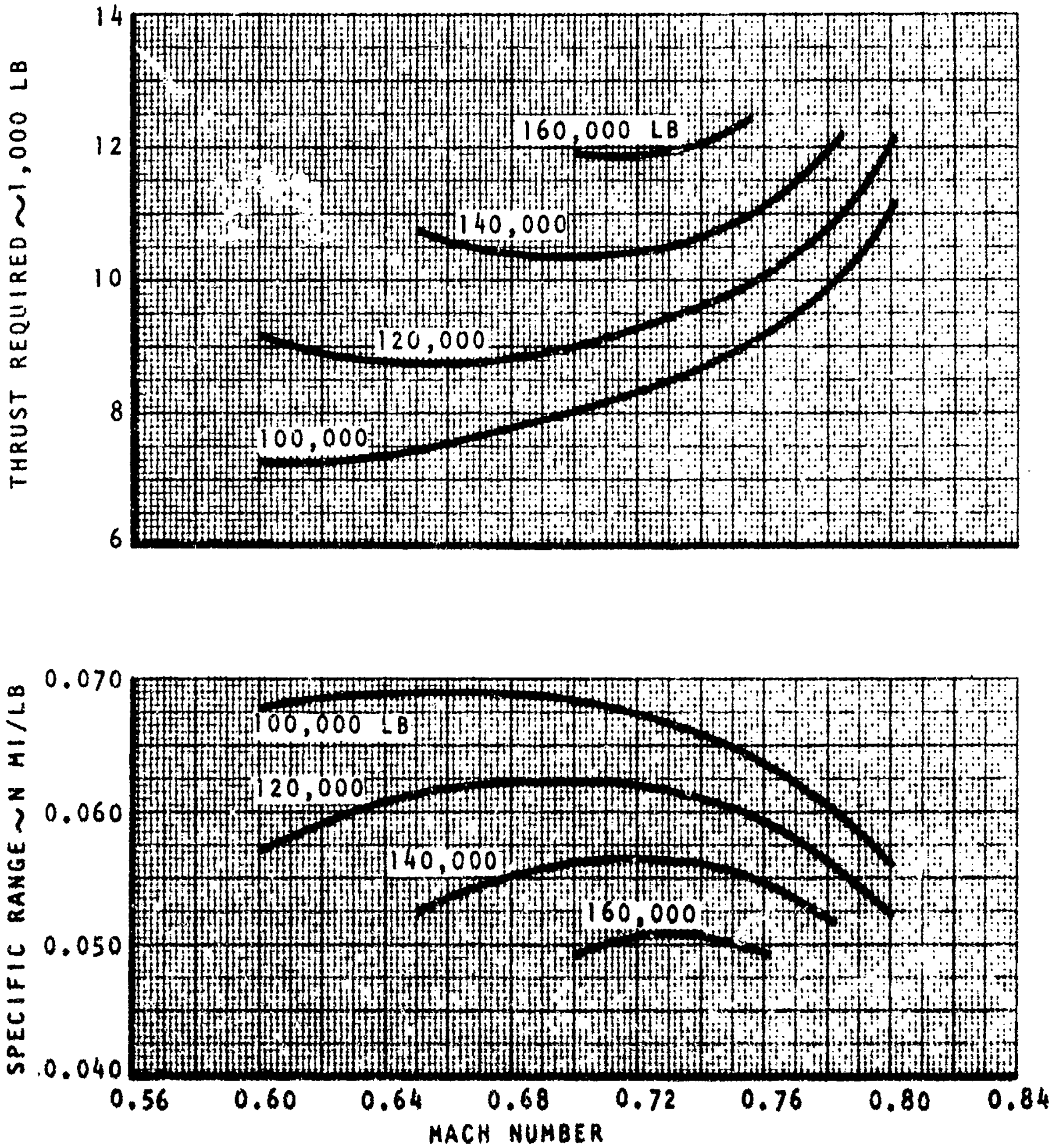


Figure 35. Thrust Required and Specific Range Cruise at 40,000 Feet

TABLE XI.

STOL EMPLOYMENT MISSION - D516-2A

Leg No.	Operation	Weight (lb)	Altitude (ft)	Mach No.	Fuel Used (lb)	Time (min)	Total Time (min)	Range (n mi)	Total Range (n mi)
	Initial weight	159,310.0							
1	Warmup, takeoff	157,064.1	0.0	0.400	2,245.9	5.000	5.000	0.0	0.0
2	Climb	152,409.7	41,232.6	0.750	4,654.4	18.806	23.806	114.44	114.44
3	Cruise	145,237.5	42,255.5	0.750	7,172.2	54.129	77.935	387.31	501.75
4	Warmup, takeoff	142,991.6	0.0	0.400	2,245.9	5.000	82.935	0.0	501.75
5	Climb	138,706.7	43,189.7	0.750	4,284.9	18.114	101.049	109.73	611.48
6	Cruise	132,049.3	44,238.1	0.750	6,657.4	54.764	155.813	391.79	1,003.26
7	Sea level loiter	127,810.6	0.0	0.268	4,238.8	30.000	185.813	0.0	1,003.26
8	Reserve	126,152.5	0.0	0.0	1,658.1	0.0	185.813	0.0	1,003.26

Total fuel (lb) = 33,157.5

TABLE XII.

STOL DEPLOYMENT MISSION-D516-2A - CRUISE AT BCA

Leg No.	Operation	Weight (lb)	Altitude (ft)	Mach No.	Fuel Used (lb)	Time (min)	Total Time (min)	Range (n mi)	Total Range (n mi)
	Initial weight	191,172.0							
1	Warmup, takeoff	188,926.1	0.0	0.400	2,245.9	5.000	5.000	0.0	0.0
2	Climb	183,429.4	36,862.1	0.750	5,496.7	20.048	25.048	121.55	121.55
3	Cruise	134,736.3	43,821.4	0.750	48,693.1	346.353	371.401	2,478.45	2,600.00
4	Sea level loiter	130,381.6	0.0	0.270	4,354.8	30.000	401.401	0.0	2,600.00
5	Reserve	127,179.7	0.0	0.0	3,201.9	0.0	401.401	0.0	2,600.00

Total fuel (lb) = 63,992.3

TABLE XIII.

STOL FERRY MISSION, D516-2A. - CRUISE AT BCA

Leg No.	Operation	Weight (lb)	Altitude (ft)	Mach No.	Fuel Used (lb)	Time (min)	Total Time (min)	Range (n mi)	Total Range (n mi)
	Initial weight	191,172.0							
1	Warmup, takeoff	188,926.1	0.0	0.400	2,245.9	5.000	5.000	0.0	0.0
2	Climb	183,429.4	36,862.1	0.750	5,496.7	20.048	25.048	121.55	121.55
3	Cruise	109,522.8	46,961.9	0.750	73,906.6	573.104	598.152	4,100.68	[illegible],222.23
4	Sea level loiter	105,823.1	0.0	0.245	3,699.6	30.000	628.153	0.0	4,222.23
5	Reserve	101,331.1	0.0	0.0	4,492.1	0.0	628.153	0.0	4,222.23
	Total fuel (lb) = 89,840.9								

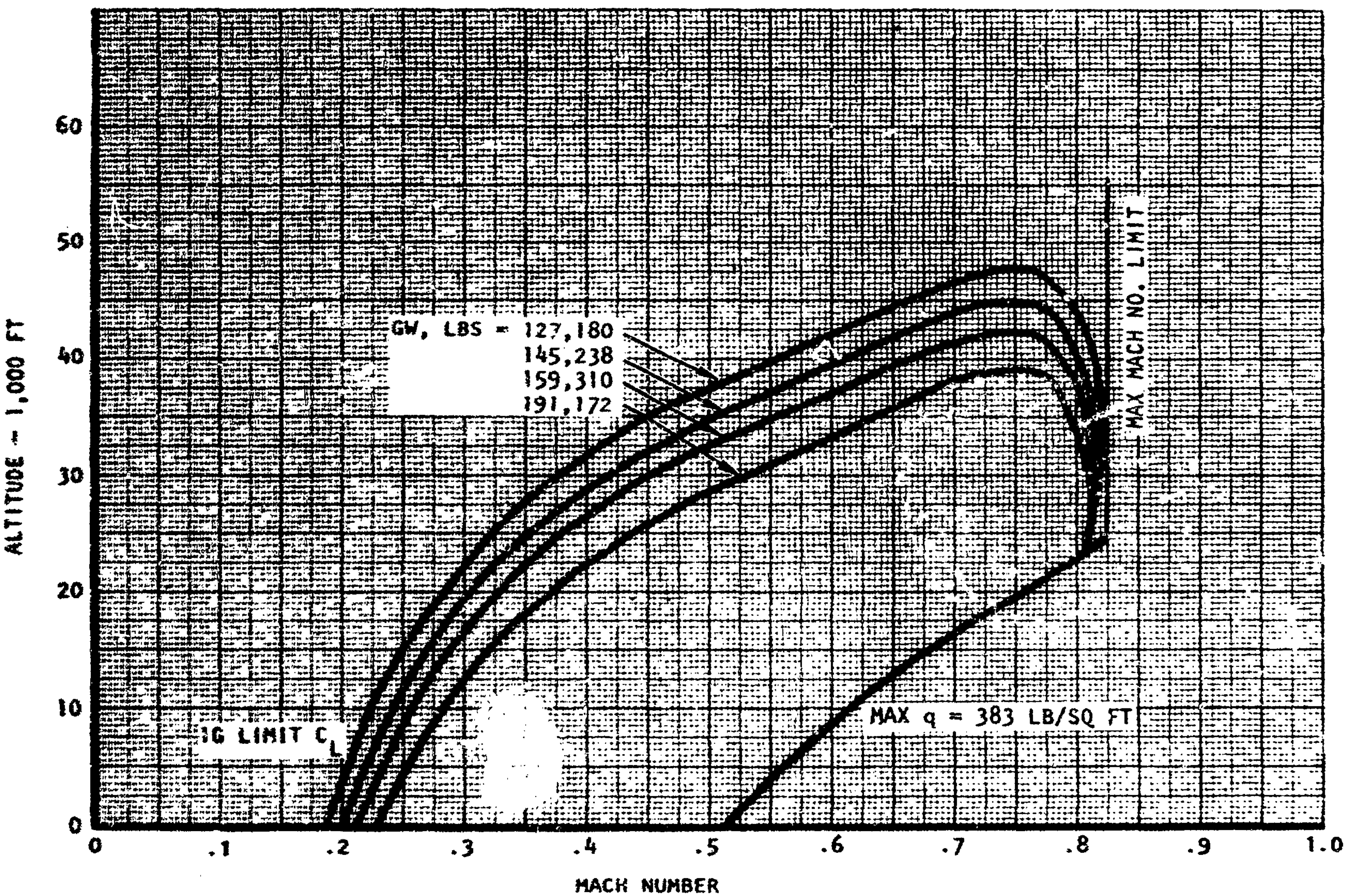


Figure 36. Speed Altitude Envelopes Intermediate Power

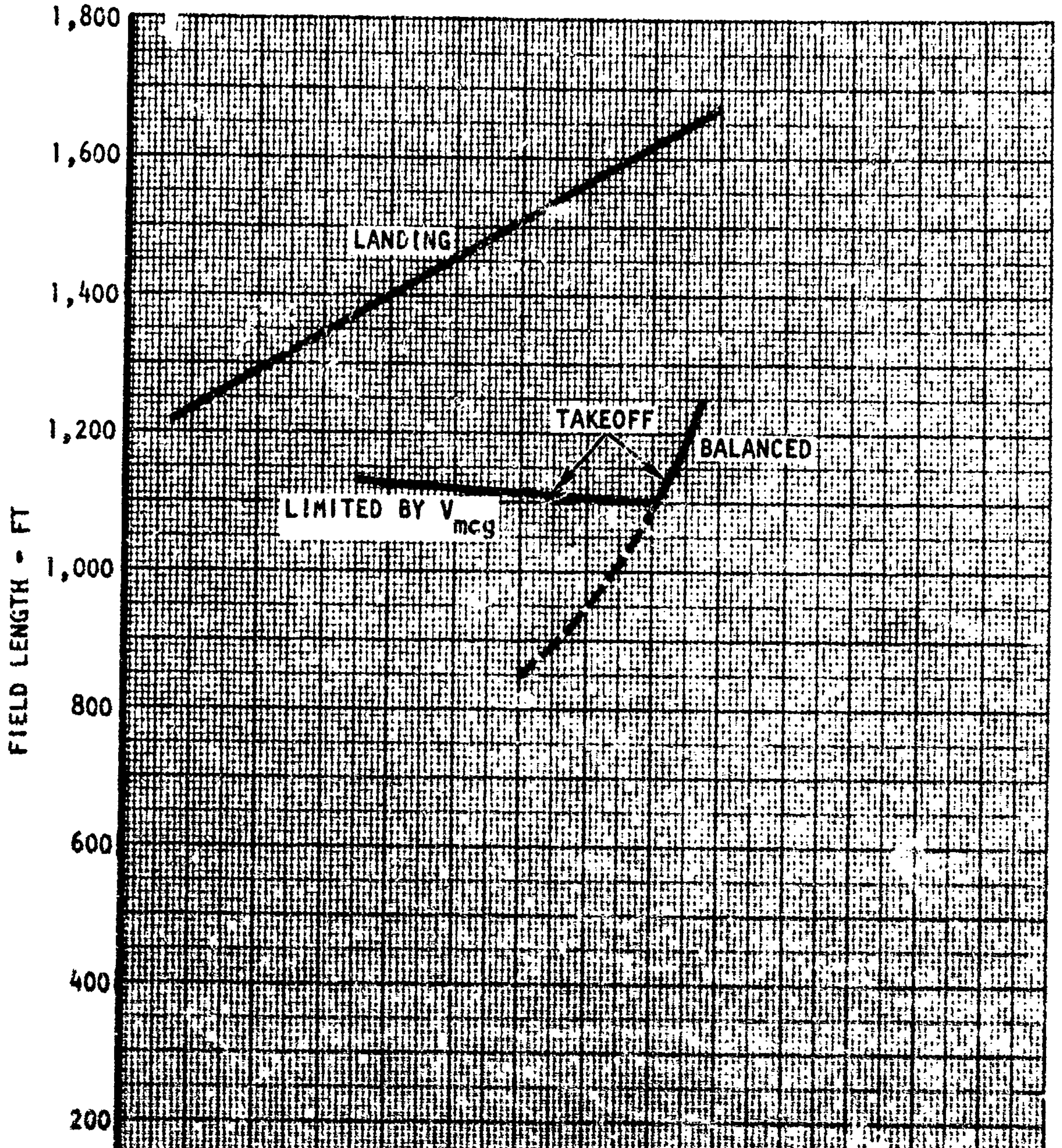


Figure 37. Short Takeoff and Landing Distance at 2500 Foot Altitude, Hot Day

V_{LO} = LIFTOFF SPEED

V_a = APPROACH SPEED

V_F = ENGINE FAILURE SPEED FOR BALANCED FIELD LENGTH

V_{mcg} = DIRECTIONAL CONTROL SPEED ON GROUND

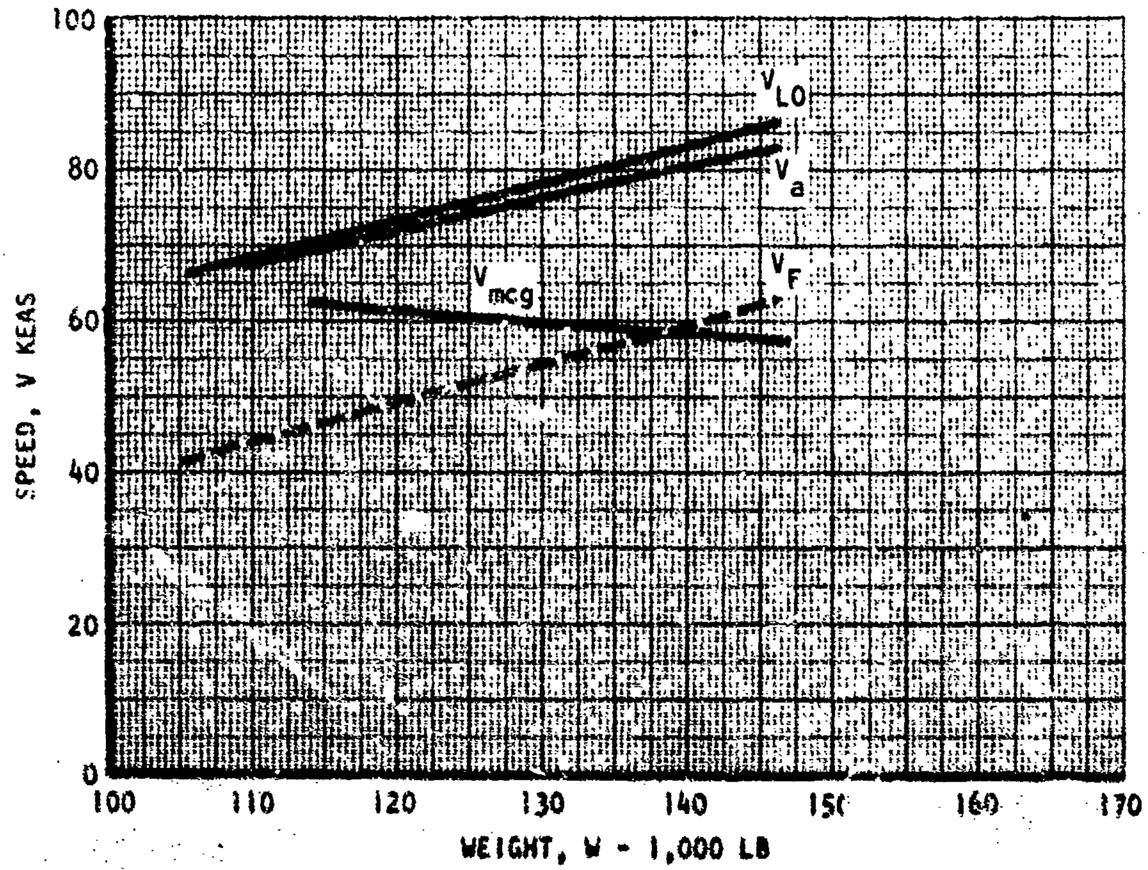


Figure 38. Critical Speeds for Takeoff and Landing

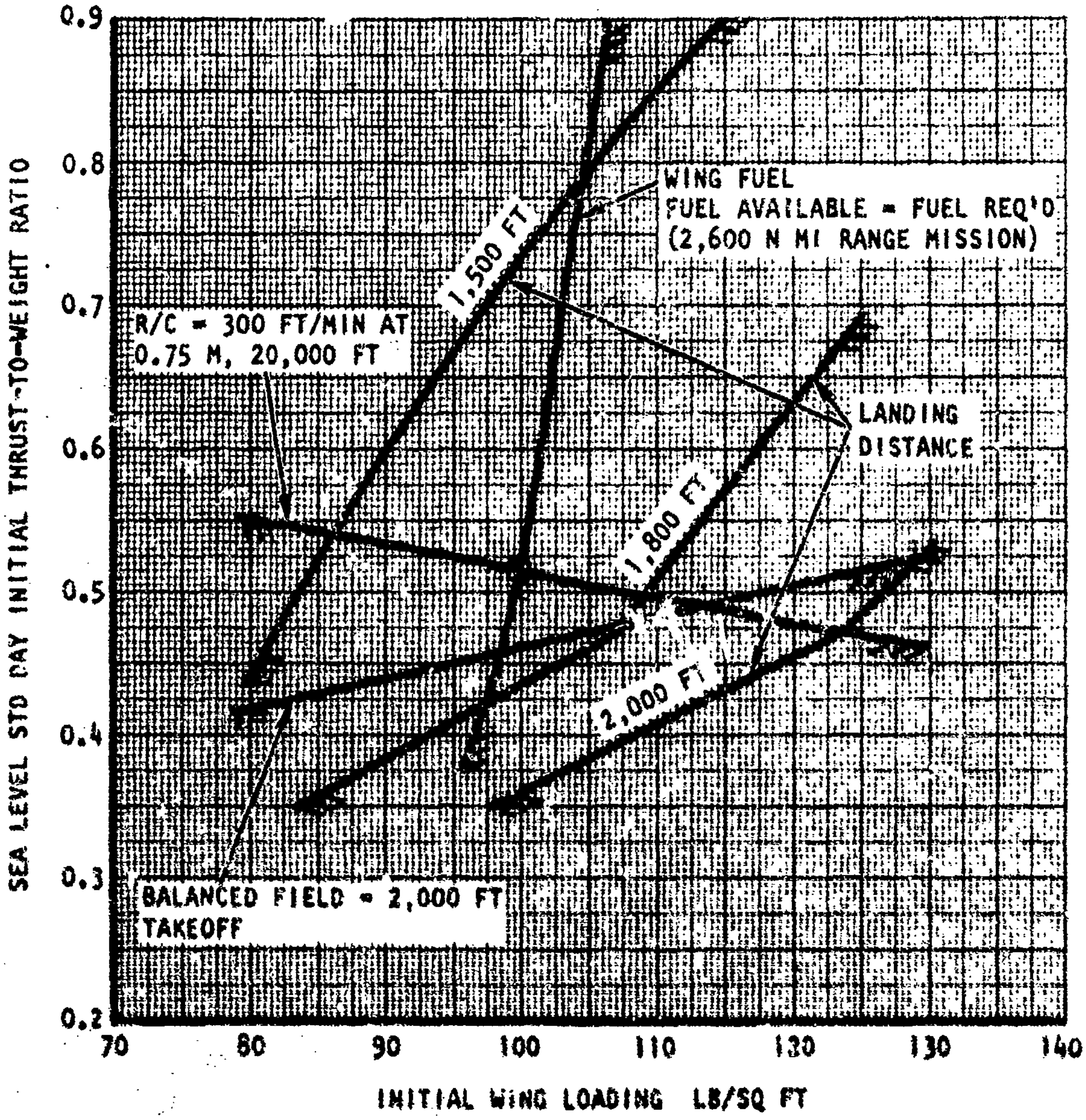


Figure 39. STOL TAI Sizing Conditions

Section VI

AERODYNAMICS

6.1 EXTERNAL AERODYNAMICS

6.1.1 CONVENTIONAL FLIGHT MODE

The external drag data presented in this section includes all external drag except that associated with the engine inlets, the drag of which is accounted for in the installed engine thrust. Included is the skin friction, upsweep, separation, and miscellaneous drags, as well as compressibility drag, tail-off drag due to lift, and trimmed drag due to lift. Camber drag is included in drag due to lift. The total drag is the sum of these.

6.1.1.1 Skin Friction Drag

The skin friction drag is presented in figure 49 together with upsweep drag and miscellaneous drags and is determined at full-scale flight conditions using the NASA skin friction program, P-7121 (reference 2), modified to incorporate a constant value of terminal C_f. This method consists of a systematic division of the airplane into components such as wings, fuselage, and nacelles in order to calculate the Reynolds number based on the proper characteristic length. The equivalent skin friction coefficient is obtained from curves for an insulated turbulent flat plate (Karman Schoenherr variation), which considers transition to occur at the leading edge and includes the Prandtl-Schlichting roughness effects. An equivalent sand height roughness of 0.0012 inch is used to approximate a mass-production painted surface. Appropriate empirical form factors are applied to account for the actual shape of the fuselage, gear fairings, and nacelles, and thickness of wings, tail, and pylons. These empirical equations are obtained from reference 3.

The Prandtl-Schlichting distributed roughness effects in the NASA program have been modified to remove the dip in the C_f versus RN curve through the transition region and thus agree with the current DATCOM method (reference 4).

6.1.1.2 Upsweep and Separation Drag

The upsweep drag is determined by relating it to the height of the end of the fuselage above the height if the fuselage were not swept up, and the length of the upswept portion of the fuselage. Experimentally determined values of upsweep drag based on the fuselage maximum cross-sectional area are

MACH	ALTITUDE	TEMP DEG R		REYNOLDS NO./FT	
0.2	0.0	518.67		1.42×10^6	
COMPONENT	S_{WET} FT^2	ℓ_{REF} FT	FORM FACTOR	C_{D_P} SMOOTH	C_{D_P} ROUGH
FUSELAGE	4,328	103.08	1.146	0.00608	0.00669
WING	2,563	15.75	1.263	.00522	.00590
HORIZONTAL	1,262	12.90	1.240	.00260	.00295
VERTICAL	918	17.47	1.300	.00189	.00214
PYLONS	295	15.50	1.203	.00057	.00065
LG PODS	873	48.75	1.063	.00126	.00141
NACELLES	1,280	16.25	1.230	.00252	.00285
AIRCRAFT	11,519			.02015	.02259

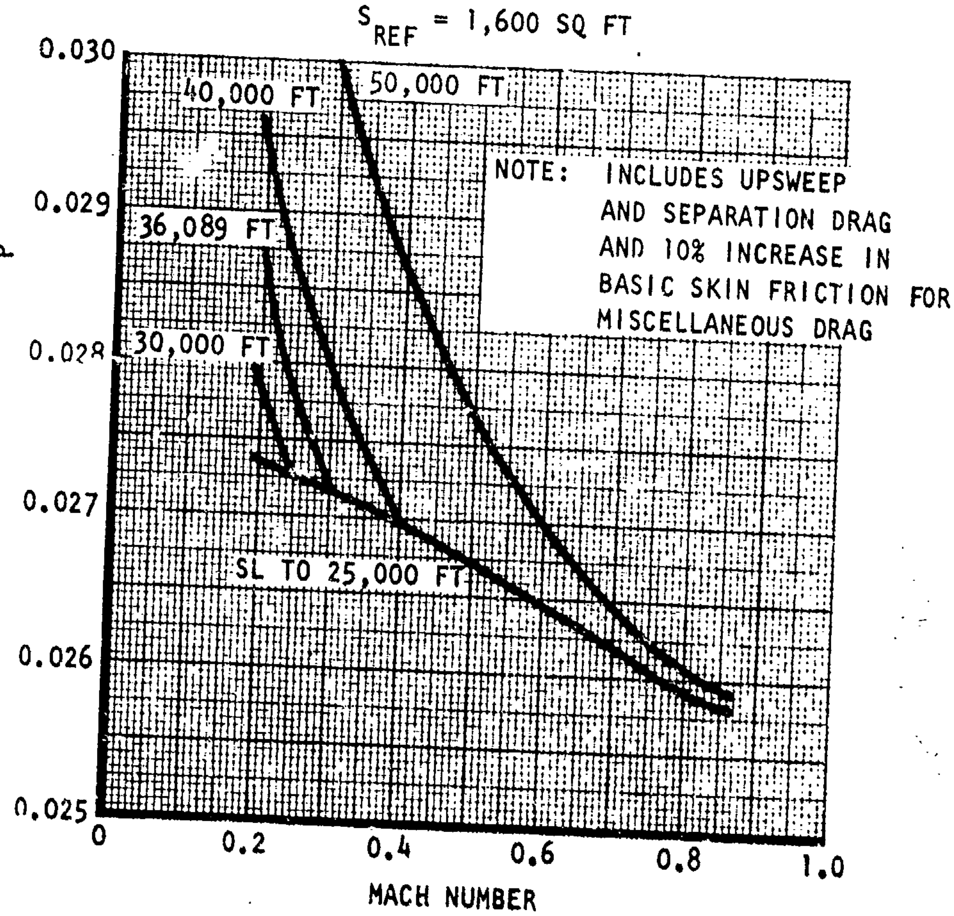


Figure 40. Skin Friction Drag

multiplied by the ratio of the maximum fuselage cross-sectional area to the reference area to obtain the upsweep drag which is presented in figure 41.

Separation drag due to flow separation from the end of the fuselage and surface actuator housings are estimated, using the base drag coefficient for boat-tailed bodies as presented in the data of page 16-4 of reference 3. The product of the base drag coefficient and ratio of separation area to reference area gives the separation drag coefficient based on the reference area. Separation is considered to occur where the rate of change of body contour is 10-degrees. The drag increment due to separation is shown in figure 41.

6.1.1.3 Miscellaneous Drag

The term miscellaneous drag is used herein to define the drag associated with small protuberances such as antennas, vents, and drains, and localized roughness such as gaps, laps, and waviness due to manufacturing irregularities on the external surface of the airplane. It includes also interference effects between aircraft components exclusive of such effects accounted for in the compressibility drag.

This drag, added to the other drag components of skin friction, upsweep, separation, and camber, gives a total drag value which correlates with parasite drag characteristics of existing aircraft. A 10 percent increase is estimated for this air vehicle.

6.1.1.4 Compressibility Drag

The subsonic compressibility drag is estimated, using the methods described in the NR Aerodynamics Manual (reference 6) for all aircraft components except the wing which is handled separately. Fineness ratios for body-type components are determined, and appropriate pressure drag coefficients are established as a function of mach number; these, in turn, are converted to drag coefficients based upon reference area. For the empennage and pylons the airfoil section, thickness ratio, aspect ratio, and sweep effects are used to evaluate the drag divergence mach number and the related compressibility drag.

For the wing, curves of C_L versus M_{DD} and compressibility drag (ΔC_{D_M}) versus ($M-M_{DD}$) figures 42 and 43 are determined from test data of air vehicles having similar wing planforms and similar wing design philosophy as discussed in paragraph 6.1.1.7. In figure 42, an M_{DD} envelope is established from the test points for the DC-8 and NR B-1 having a low ratio of local to freestream mach number due to fuselage perturbation velocities.

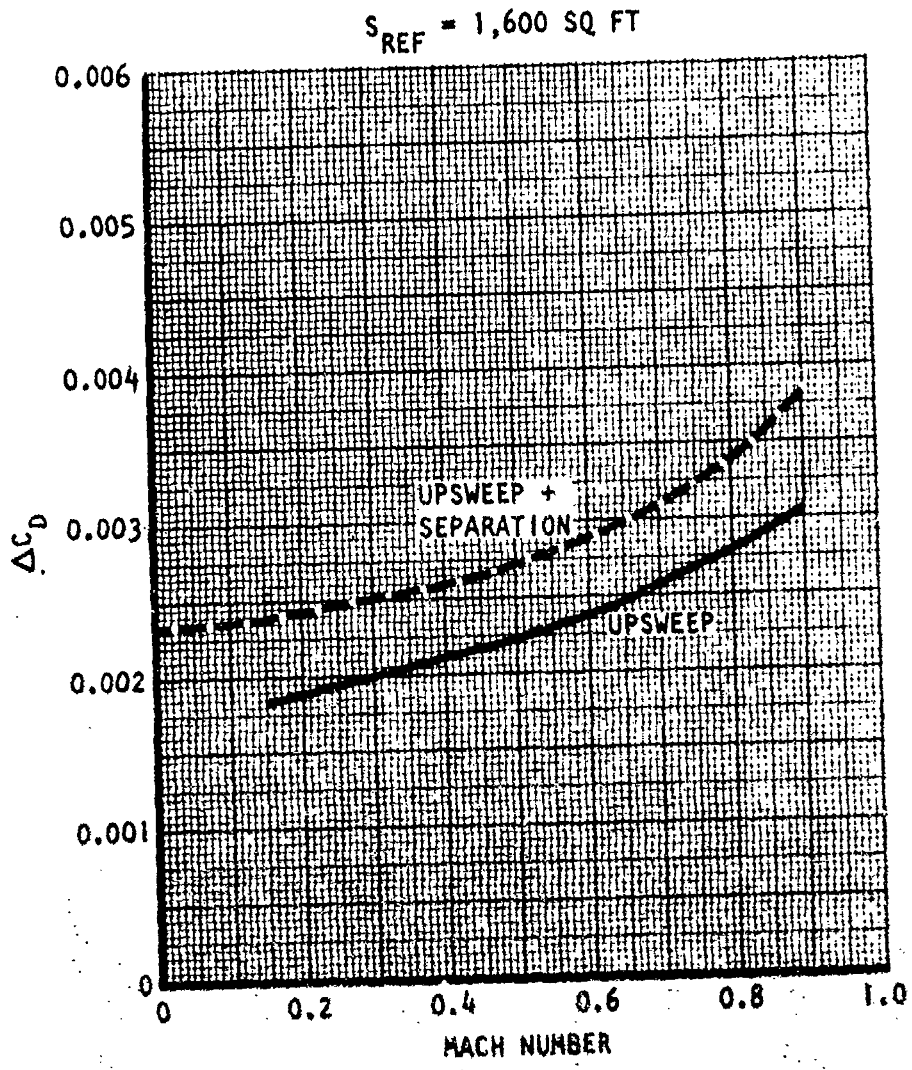


Figure 41. Upsweep and Separation Drag

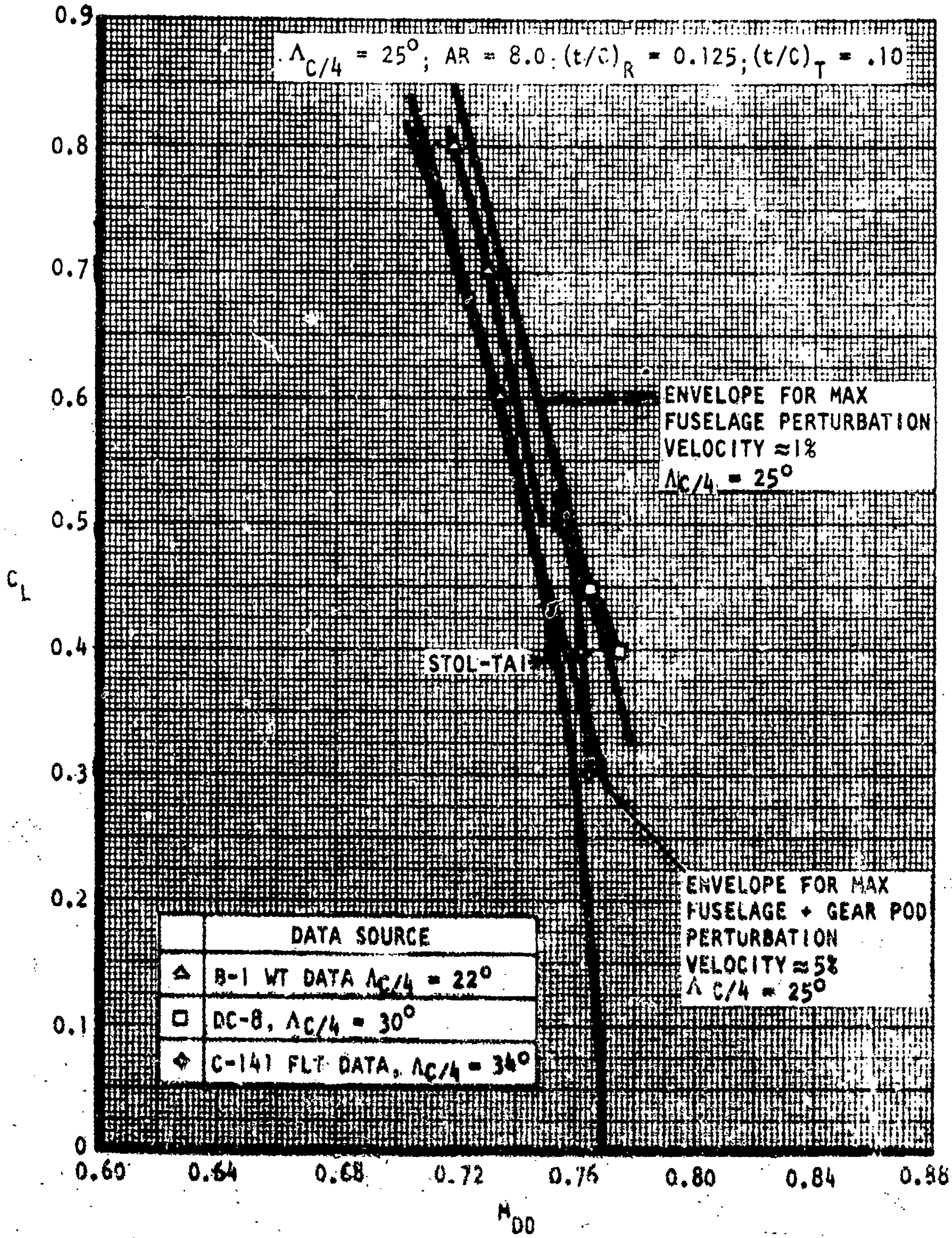


Figure 42. Drag Divergence Mach Number Versus C_L

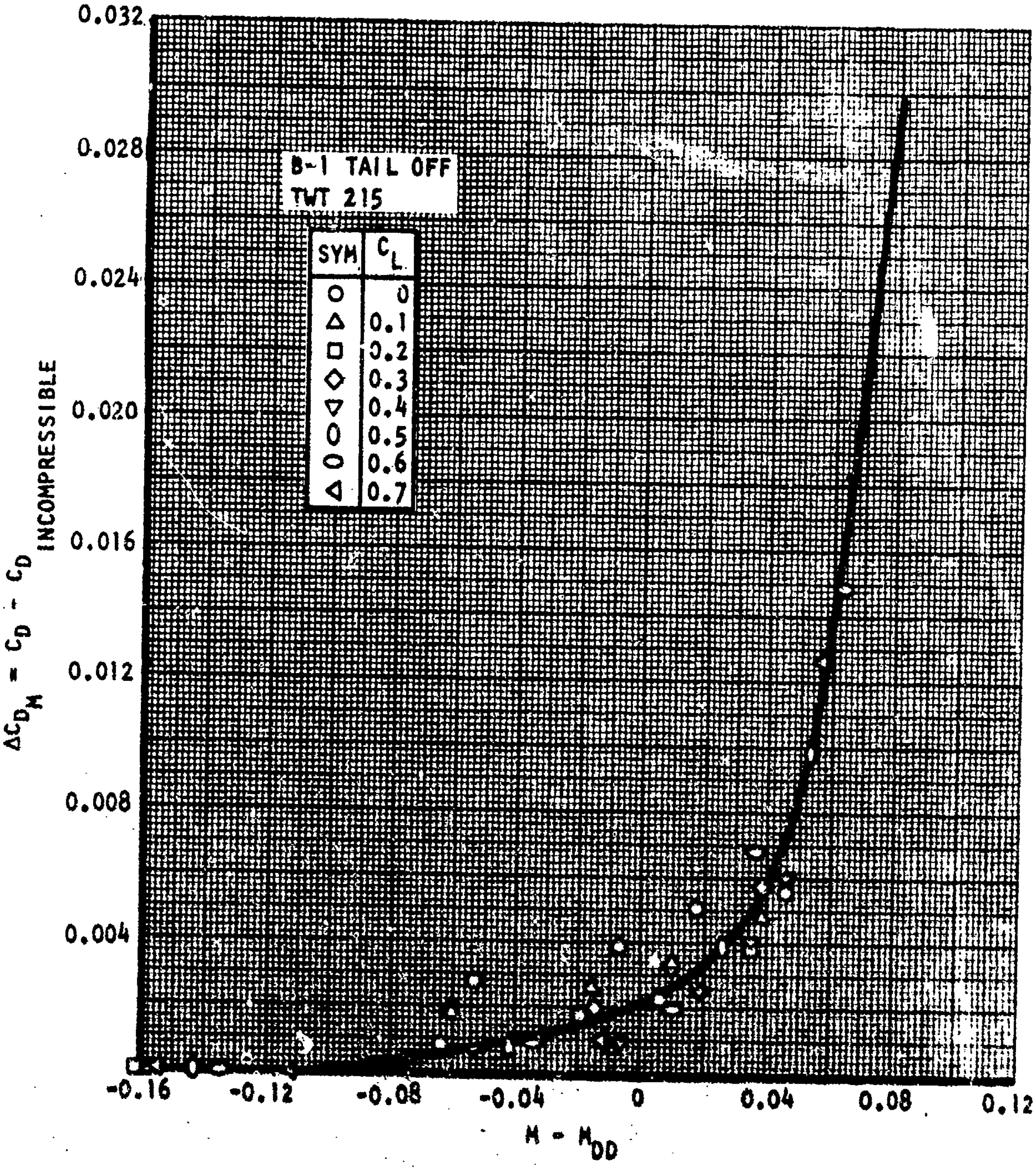


Figure 43. ΔC_{D_M} vs M-M_{DD}

The envelope is made taking into account the various sweep angles of these aircraft. To account for the fuselage and gear pod effects on the M_{DD} of the present aircraft, the data of figure 44 are used to modify the envelope established from the test data. For the present aircraft, figure 44 shows a maximum of 5 percent increase in local mach number as contrasted to, for example, 1 percent increase estimated for the aforementioned test data, resulting in a new estimated envelope shown in figure 42. The wing used has drag divergence mach numbers consistent with this new envelope, being tangent with it at high cruise lift coefficients, as also shown in figure 42.

By means of a computer program (reference 5) the wing compressibility drag is calculated for tail-off lift coefficients using figures 42 and 43 and includes the compressibility drag due to lift. The total compressibility drag, wing plus other components, is presented as a function of mach number and trimmed lift coefficient in figure 45.

6.1.1.5 Drag Due to Lift

The tail-off incompressible drag due to lift (figure 46) is calculated by use of the efficiency factor curve of figure 47. This curve was generated from experimental test data and by wing design methods used by NR as explained further later on in this section. As is noted, the maximum e-values of some test data occur at high lift coefficients. It was found that a lift coefficient of approximately 0.45, a relatively high value for e can be obtained by reducing the camber, but this reduces the e-values at higher lift coefficients. The lowest curve is used, although the design lift coefficient is approximately $C_L = 0.65$. This yields a nominal value of $e = 0.80$ and leaves a margin for possible contingencies.

6.1.1.6 Trimmed Drag Due to Lift

The incompressible trimmed drag due to lift is determined by use of the NR trimmed drag program (reference 5). It is a slight function of mach number and presented as a function of trimmed lift coefficient in figure 48.

6.1.1.7 Wing Design

The development of the wing characteristics is accomplished using methods derived at NR which optimize the aerodynamic characteristics for the particular mission requirements (reference 7, 8, and 9). This wing optimization process includes the effects of mach number, viscosity, and leading edge suction.

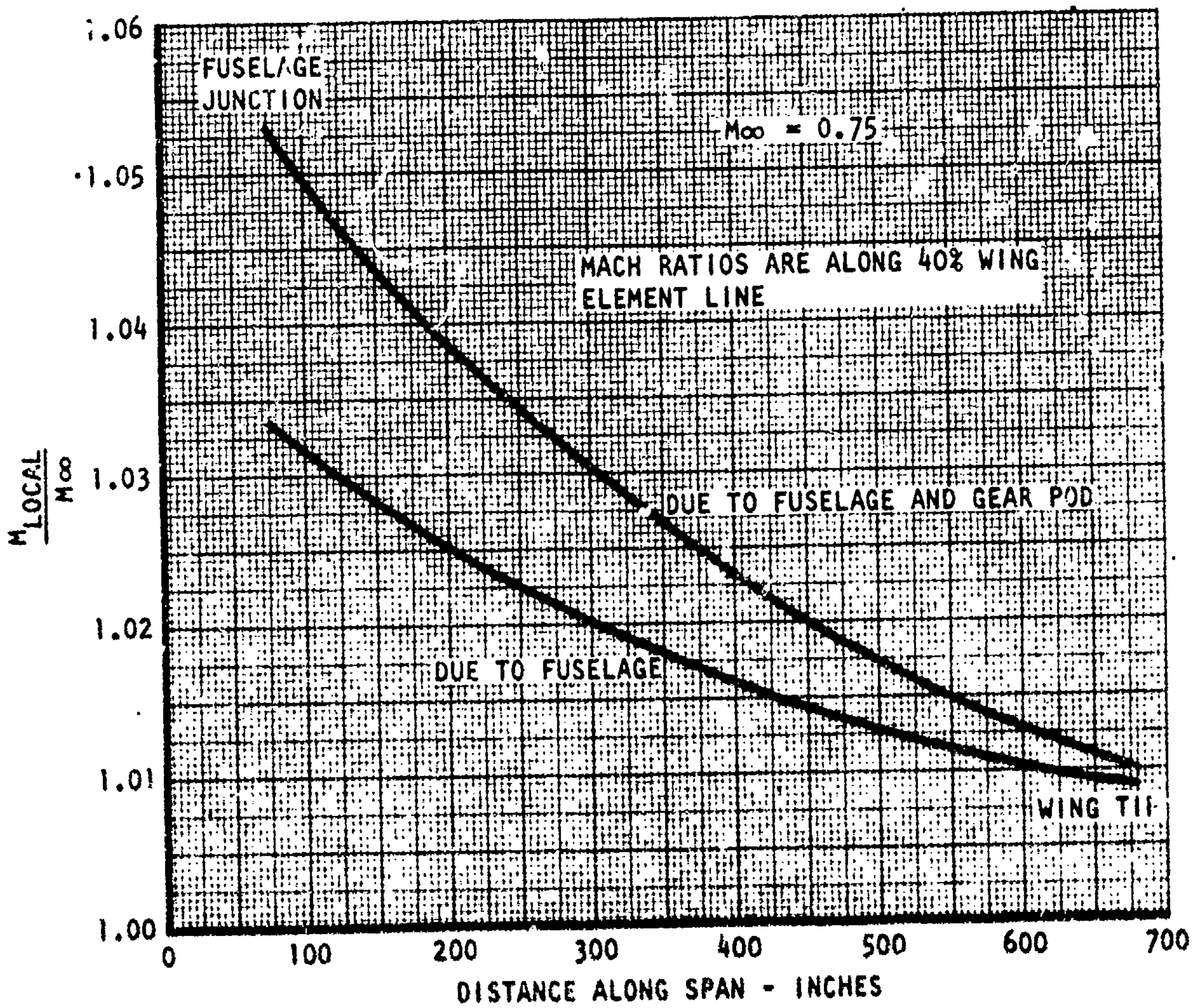


Figure 44. Fuselage and Gearpod Perturbation Effects on Local to Free Stream Mach Number Ratio

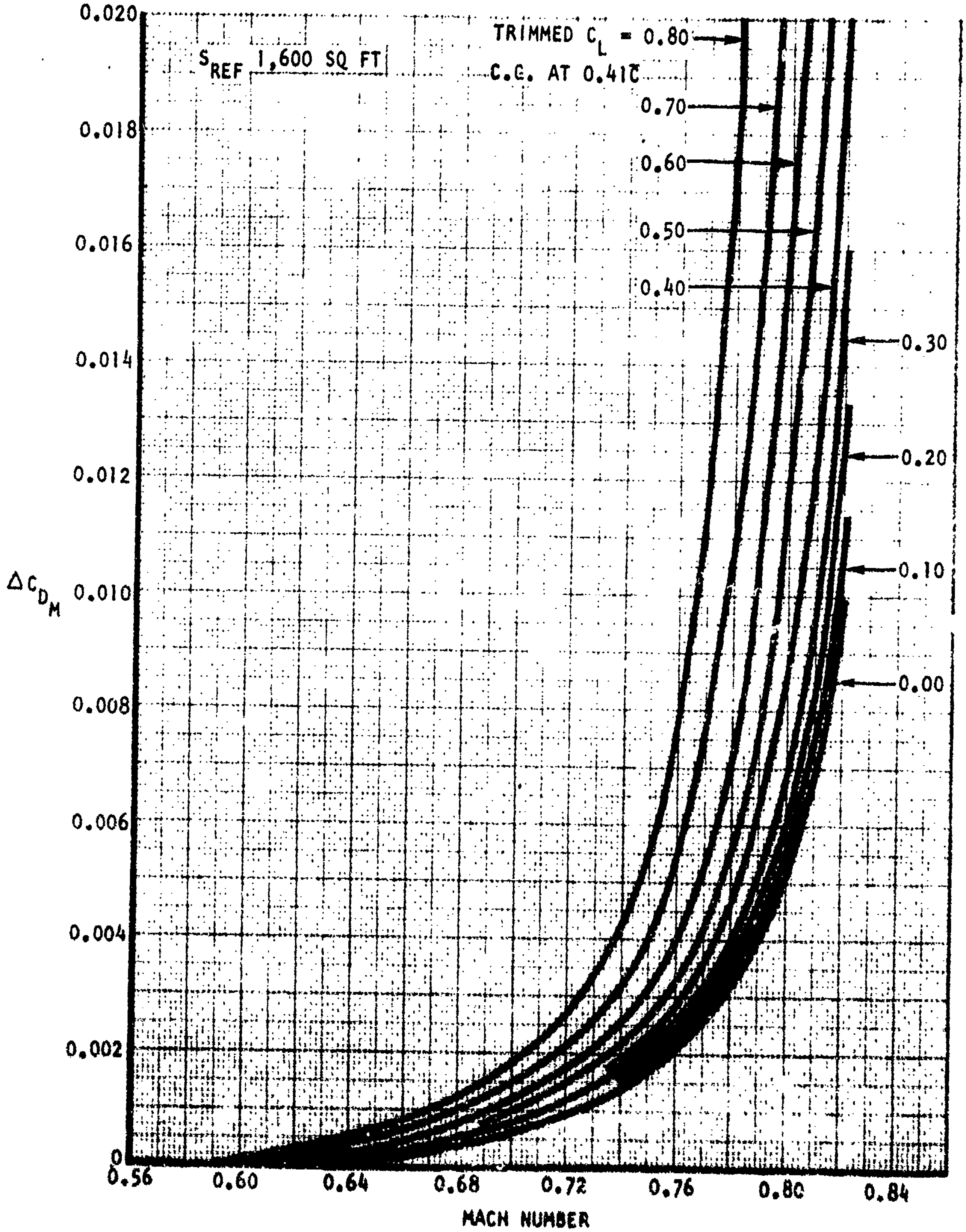


Figure 45. Compressibility Drag

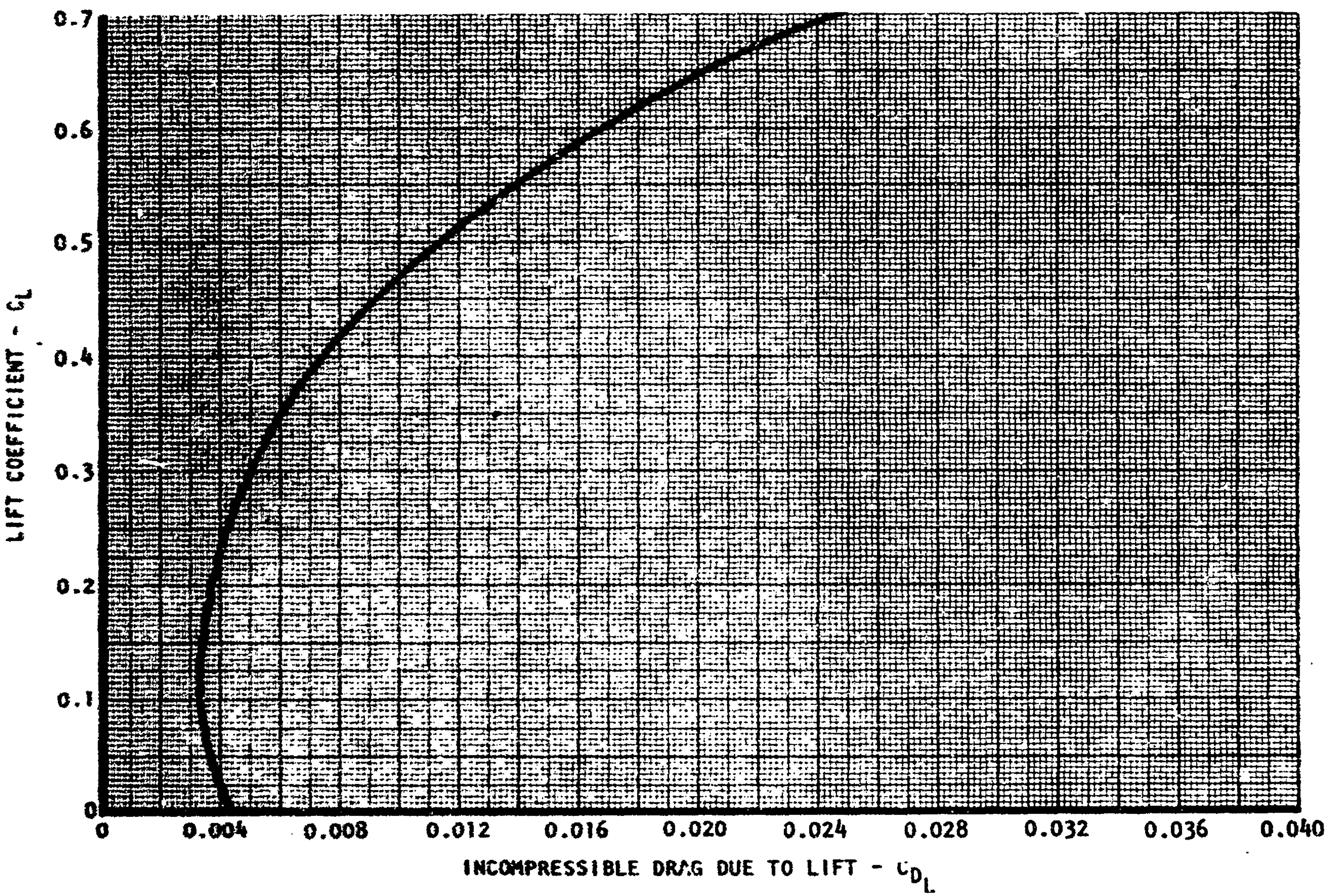


Figure 46. Incompressible Drag Due to Lift

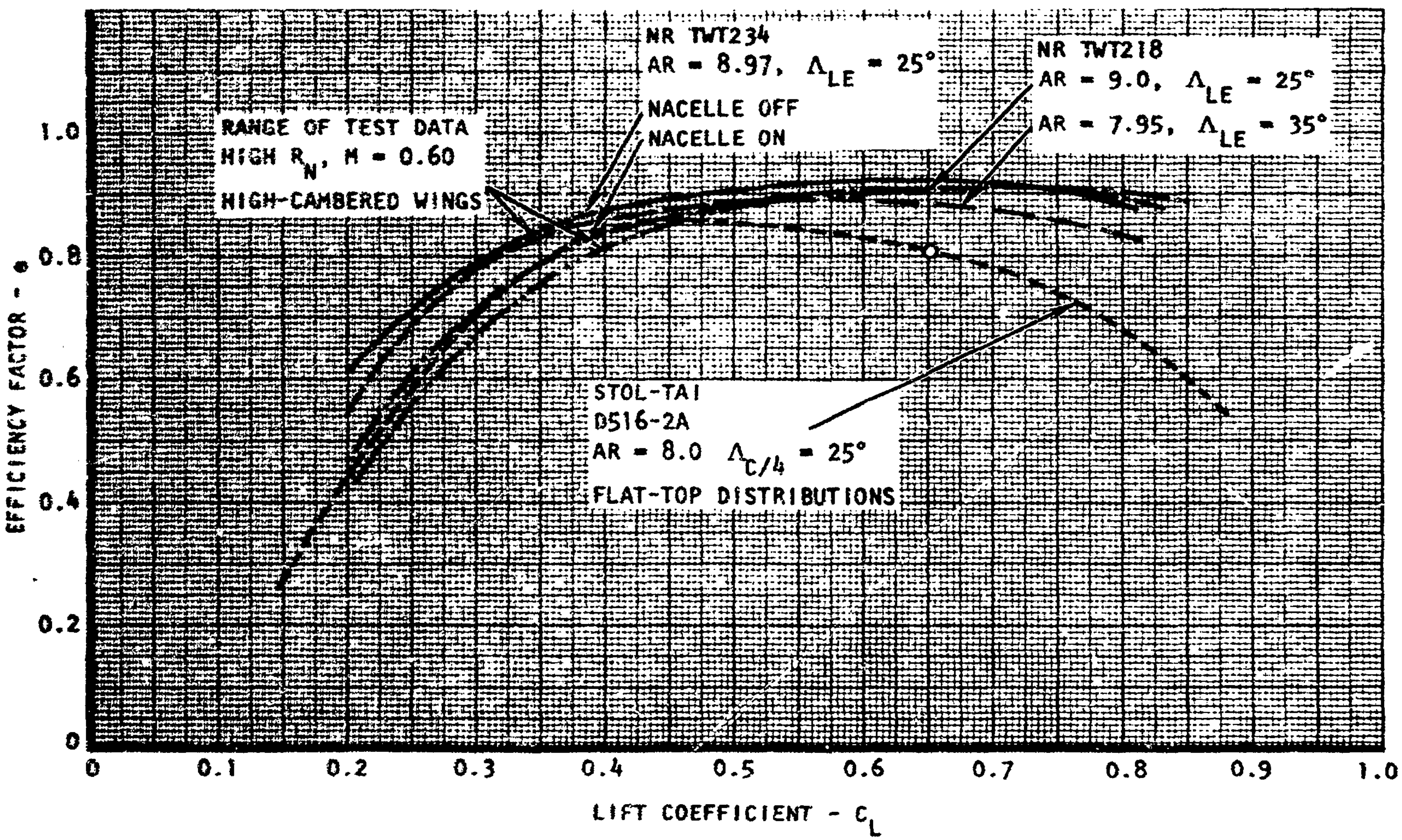


Figure 47. Incompressible Tail-Off Efficiency Factor

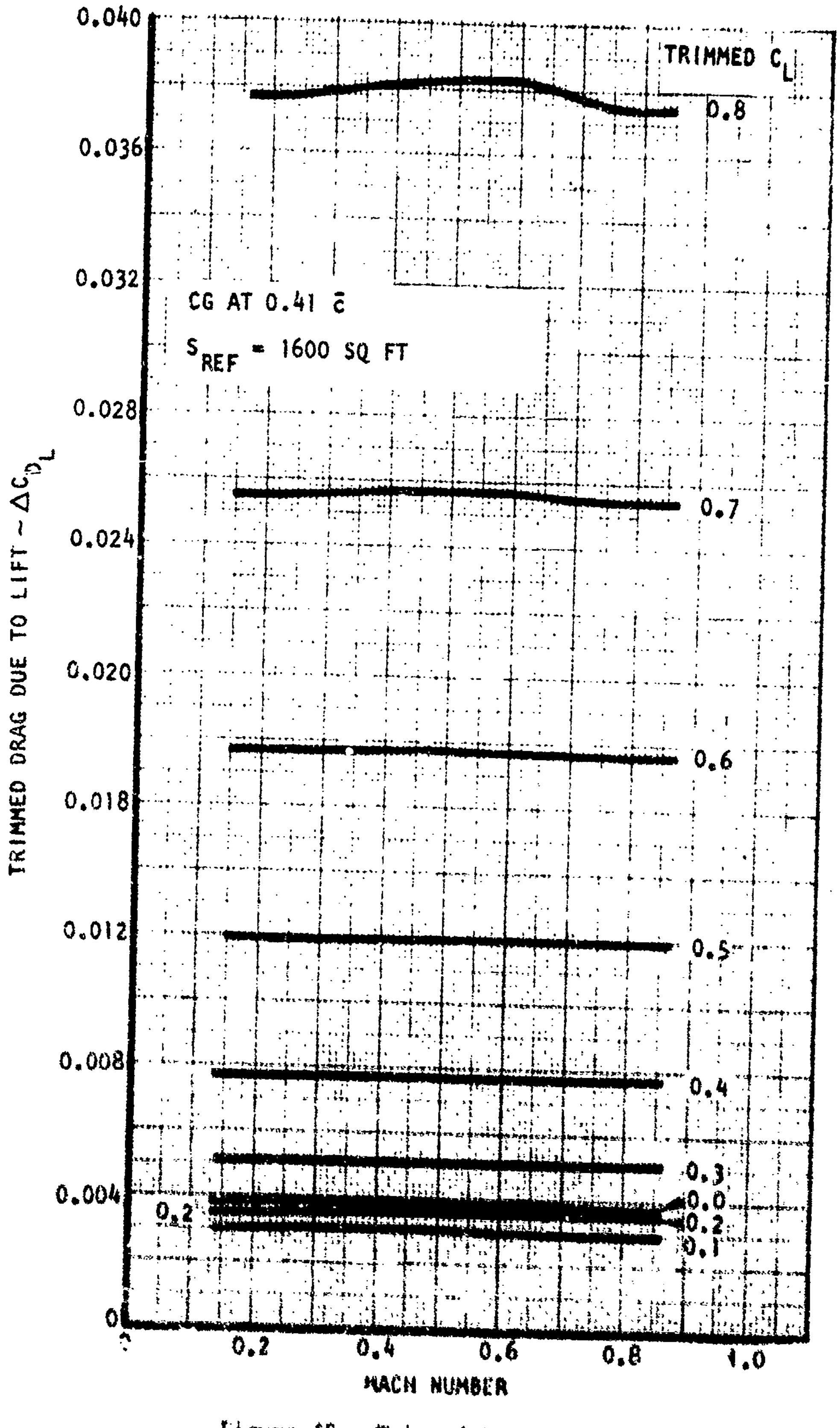


Figure 48. Trimmed Drag Due to Lift

Using an inverse vortex-lattice lifting surface program, the desired net spanwise and chordwise pressure distributions are stipulated and the required camber shape is obtained to support the load. The chordwise distribution chosen is one which solves for 100 percent suction recovery and is obtained by forcing the leading edge singularity to zero across the complete span. The percent suction actually available in viscous flow is determined through correlation of wind tunnel data obtained on similar "tuned" wing designs.

Figure 49 presents the percent suction corresponding to the test data of figure 47 and the suction obtained for the estimated efficiency factor of the STOL-TAI. A maximum suction of 95 percent of 0.45 lift coefficient for the STOL-TAI correlates well with the maximum values obtained for the test data.

To assure adequate drag divergence mach number levels, the determination of the chordwise thickness distributions is made by stipulating the upper surface chordwise distribution and the airfoil crest mach number. The airfoil thickness distribution is then determined by inputing the desired incremental velocity due to thickness and solving for the resulting thickness distribution.

6.1.2 STOL FLIGHT MODE

Power-on aerodynamic data used for the determination of the takeoff and landing performance in the STOL mode are based on the wind tunnel data obtained from the NR 7 percent STOL transport model (GELAC, LSWT-090). The intake momentum drag measured during calibration runs in the wind tunnel showed levels that are of the same order as those of the transport design so that no intake drag correction is applied to these data.

The STOL performance depends largely on the lift characteristics which are derived from these test data for the selected full-span, double-slotted flap. Maximum trimmed lift data for all engines operating are shown in figure 50 and for one engine failed in figure 51. The data in both figures are presented for various flap angles, and given as a function of the inverse of the blowing coefficient. In each figure, the maximum lift is defined as the lift existing at an angle of attack of 18 degrees. At higher angles of attack, the critical engine failed rolling moment becomes too large because of asymmetric wing stall.

Lateral trimming is performed by determining the rolling moment due to engine failure, and by applying lift loss due to roll control. The rolling moment due to rudder deflection needed to trim the yawing moment is included herein. Longitudinal trimming is performed in a conventional manner. An example of the lift loss due to roll control is presented in figure 52.

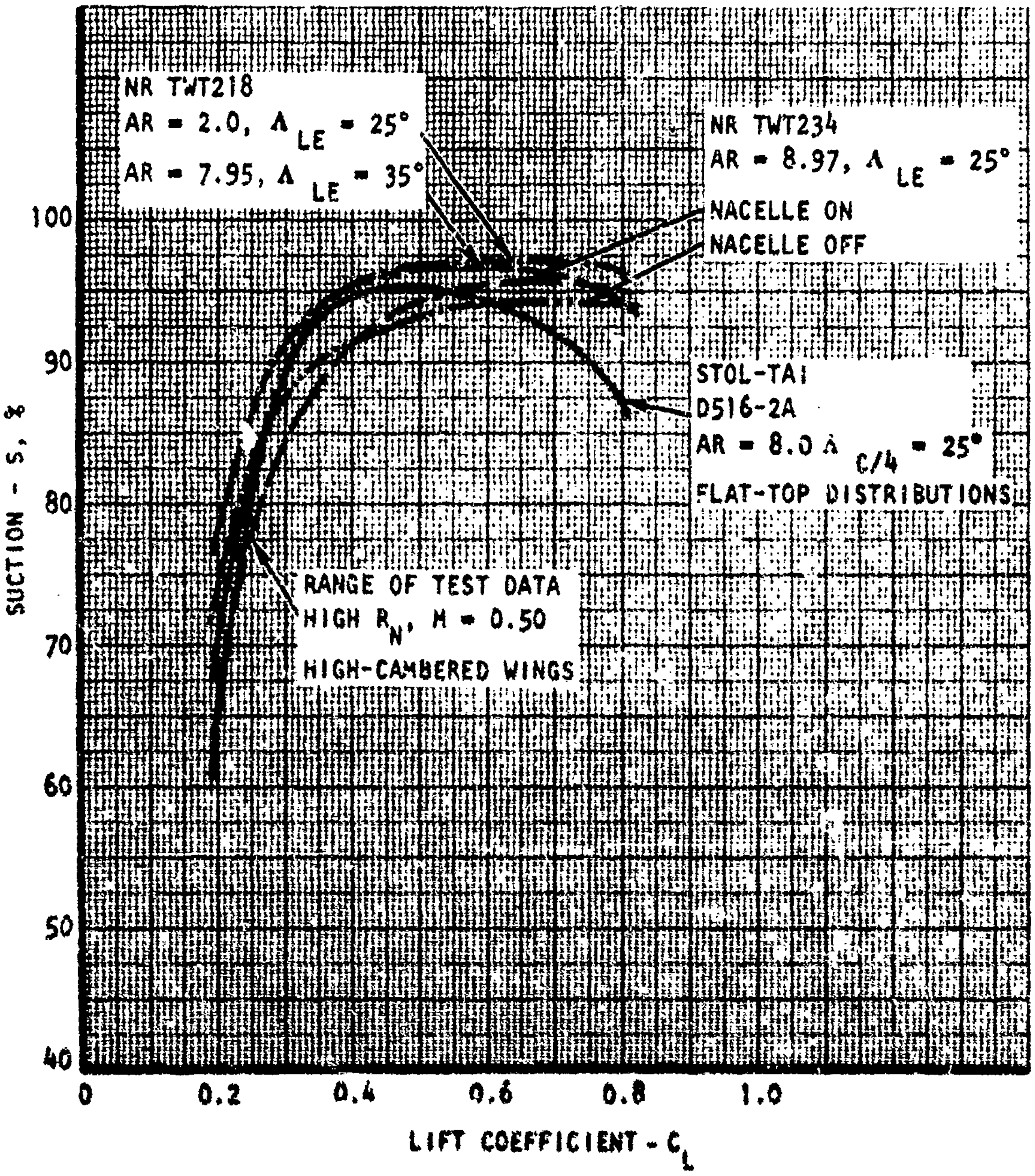


Figure 49. Leading Edge Suction

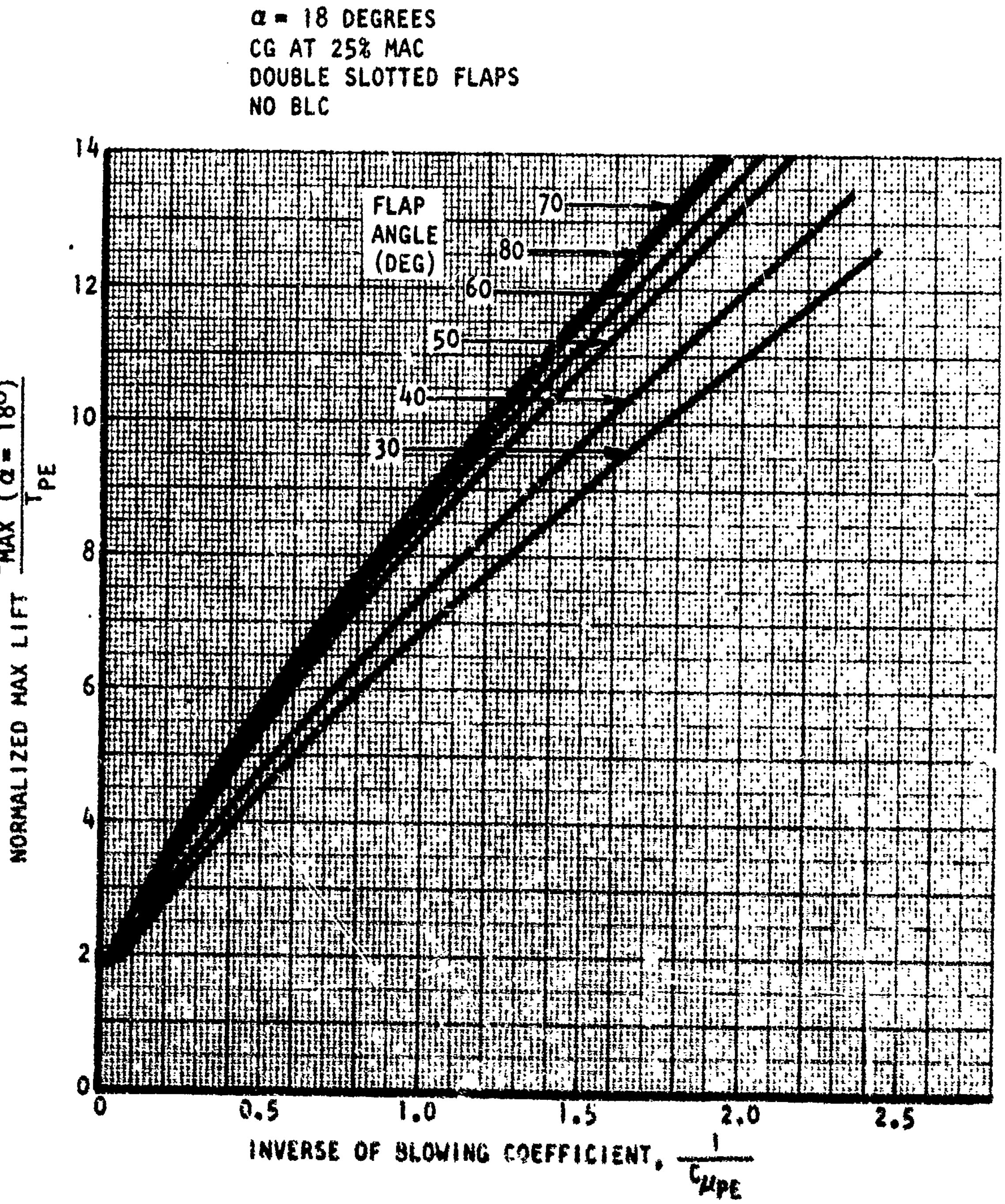


Figure 50. Maximum Trimmed Lift with All Engines Operating

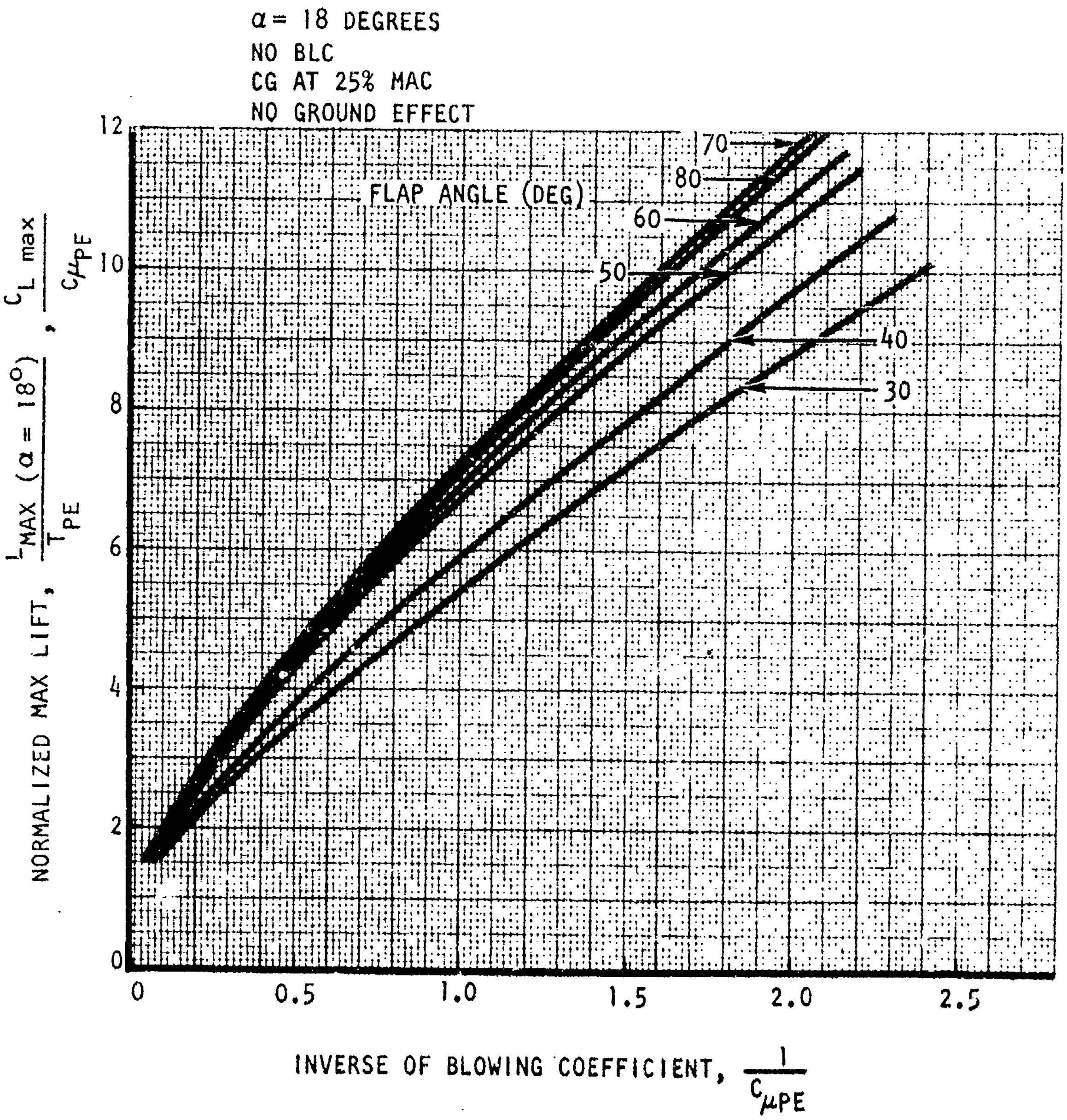


Figure 51. Maximum Trimmed Lift for Double-Slotted Flaps and Critical Engine Failed

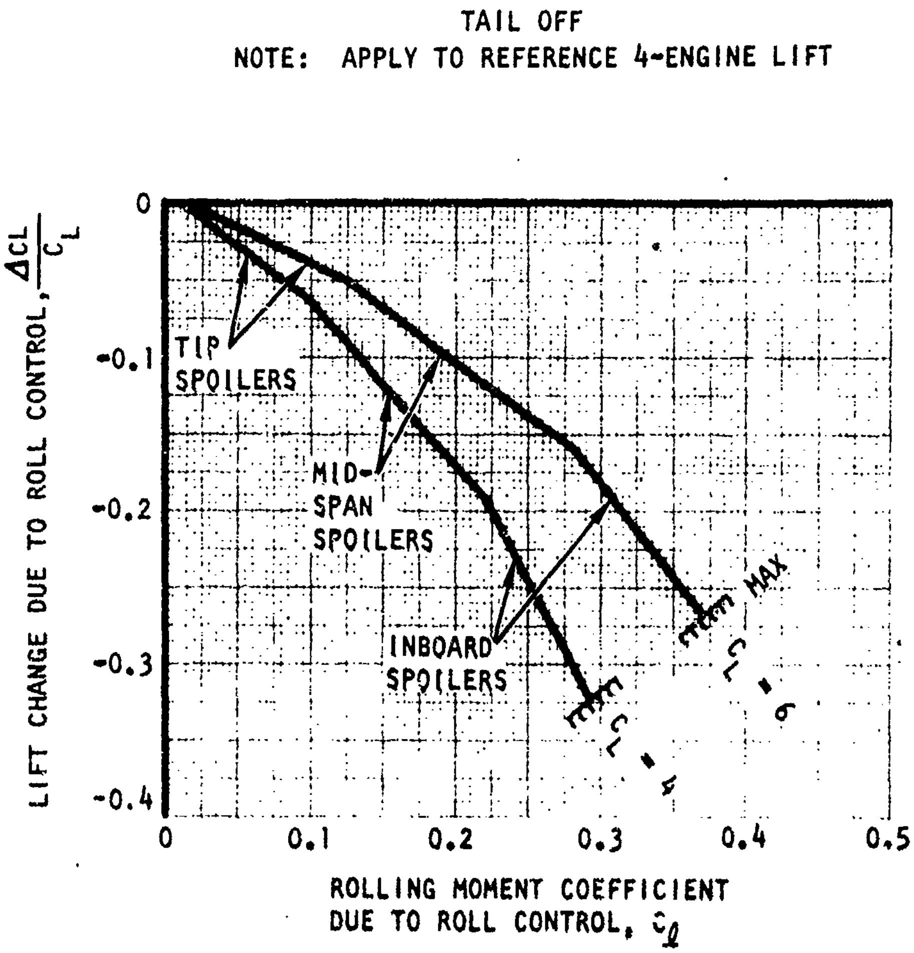


Figure 52. Lift Change Due to Roll Control

Cross plots of maximum trimmed lift are presented as a function of flap angle, in figures 53 and 54, each for a certain blowing coefficient. In these figures, takeoff maneuver margins and speed margins are applied to obtain the lift level suitable for the takeoff or landing condition.

Lift characteristics in ground effect are also included in these figures. The maximum lift in ground effect is obtained using figures 55 and 56 for the ratio of lift-in-ground effect to that out-of-ground effect, keeping the angle-of-attack constant. These figures were derived from unpublished NASA wind tunnel tests at constant angles of attack (LWP 987).

In addition to the effect on lift at a constant angle of attack, the ground proximity also reduces the magnitude of the stall angle of attack. This effect is shown in figure 57, and in the absence of test data, was computed from existing equations valid for conventional aerodynamics. These are primarily based on the effect of image trailing vortices. It was assumed that the effect of these vortices on the angle-of-attack change is identical between STOL aircraft with externally blown flaps and conventional aircraft.

The total ground effect is then determined first by determining the change in α_{max} from figure 57, then by obtaining the lift out-of-ground effect for this angle, and then applying figures 55 and 56 to this lift.

Figure 53 also shows the lifting capability at which a climb angle of $\gamma = 3$ degrees can be maintained with the critical engine failed. Note that with increase of flap angle this lift decreases. This is due to the decrease in the lift/drag ratio, and for a drag level consistent with a constant γ the lift must then decrease.

Lift and drag characteristics needed to determine the lifting capability at a flight path angle of $\gamma = 3$ degrees are presented in figure 58 as an example. The curve shows first the effect of engine failure on the lift/drag relation in the untrimmed condition, and then the effect of trimming in roll, yaw, and pitch. The trimming in roll reduces the drag at constant angle of attack, because of the reduced lift. Also, trimming in pitch reduces the drag somewhat, because a download on the tail is required; this download is inclined forward, because of the rather large downwash.

The takeoff condition is obtained taking into account all required safety margins in terms of speed and maneuver capability maintaining at least a three-degree climb angle with the critical engine failed.

The best flap angle is the one where the highest lifting capability is obtained that meets the safety margins as well as the climb conditions as shown in figure 53. This lifting capability is plotted in figure 59 as $C_L/C\mu_{PE}$ versus $1/C\mu_{PE}$, and the associated flap angles are

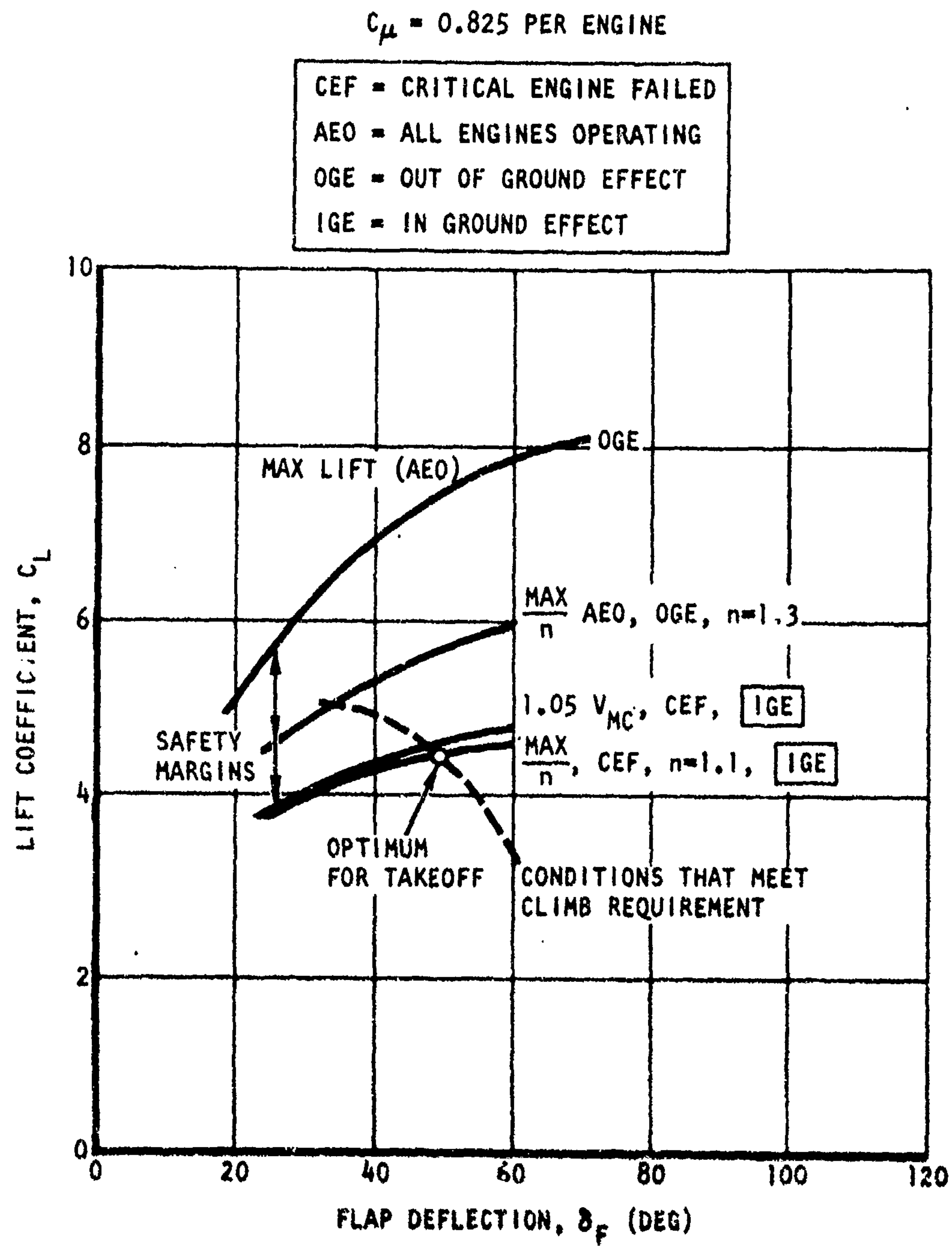


Figure 53. Lift Characteristics for Takeoff Conditions

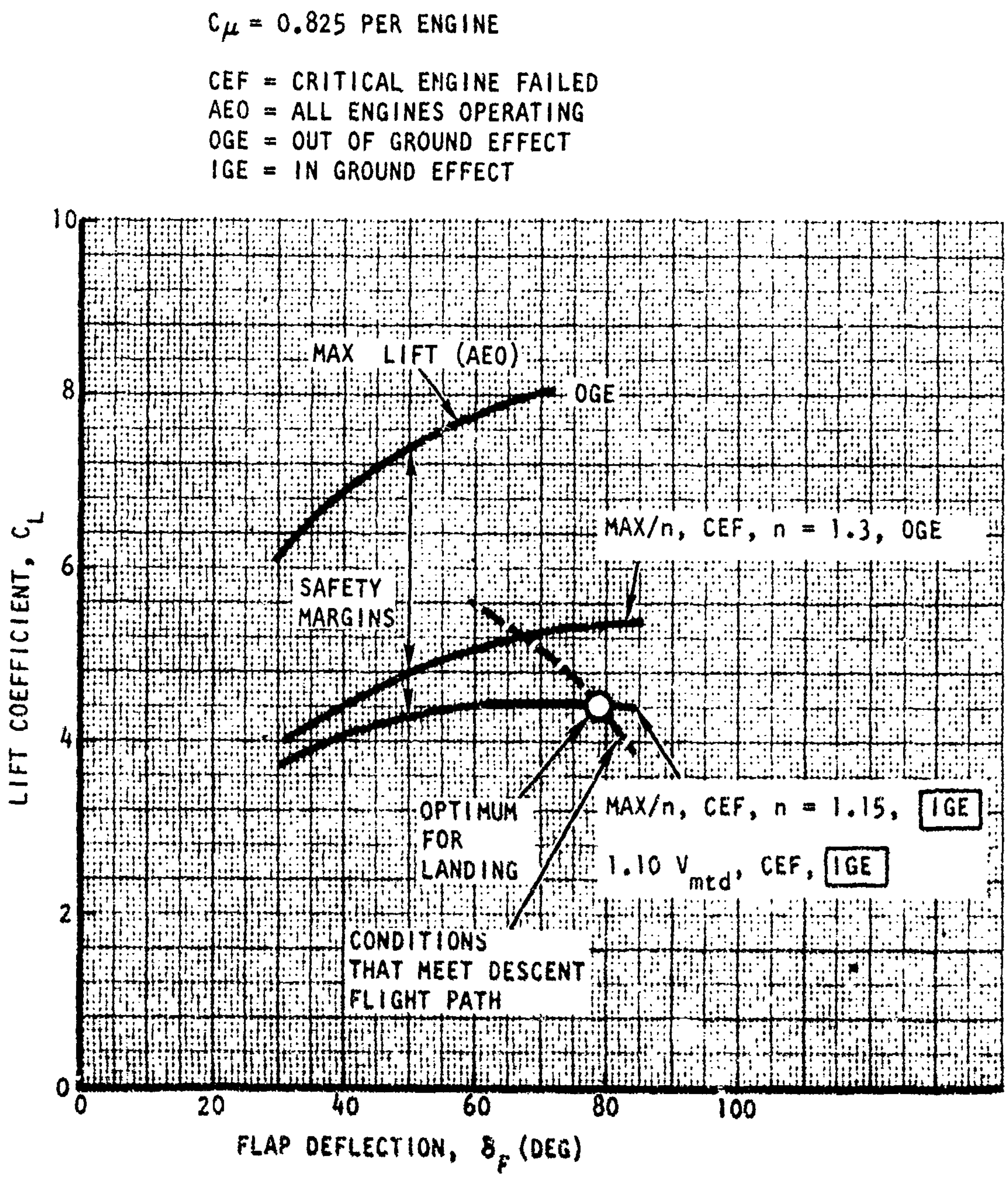


Figure 54. Lift Characteristics for Landing Conditions

δ_F = 35 DEGREES

TAIL OFF

CONSTANT ANGLES OF ATTACK BELOW $C_{L_{MAX}}$

ALL ENGINES OPERATING

REF: LWP-987

h = HEIGHT OF $\bar{c}$/4 ABOVE GROUND

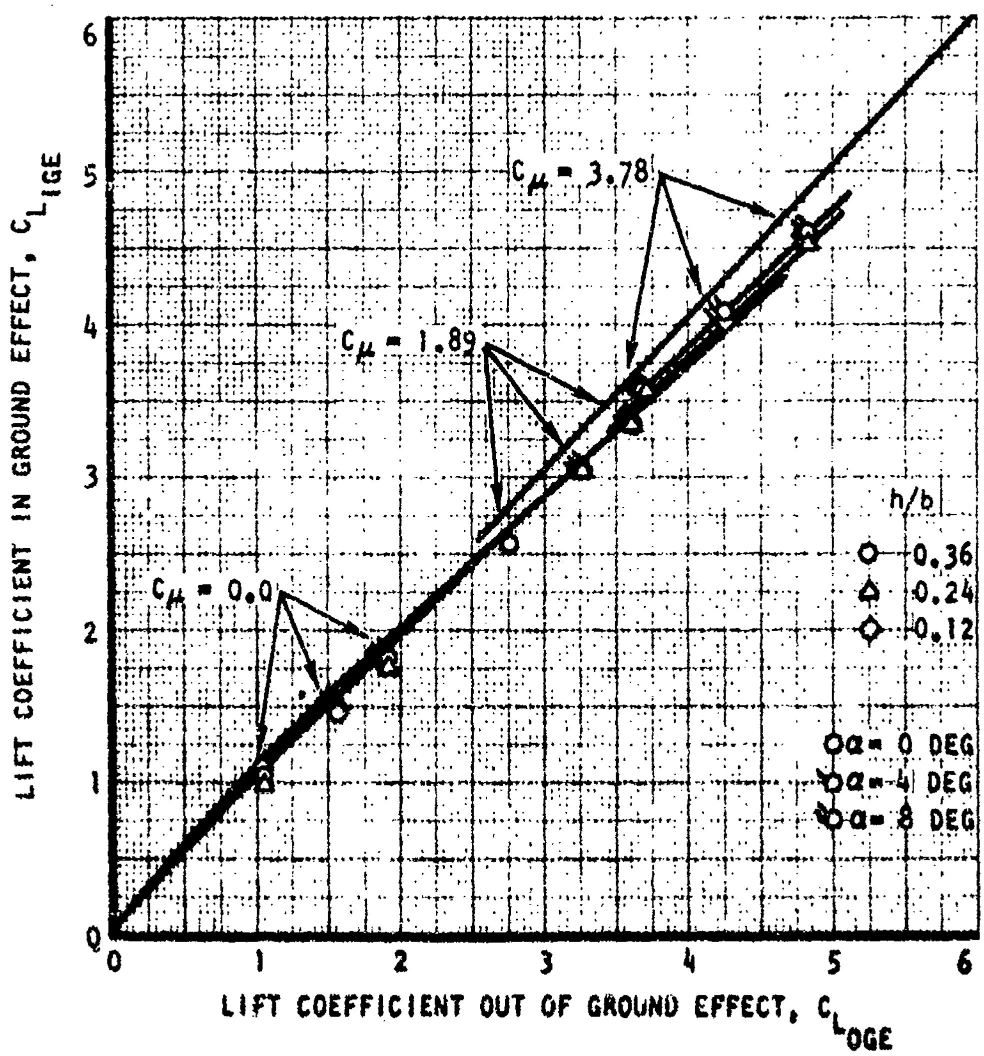


Figure 55. Ground Effect on Lift for Externally Blown Flaps

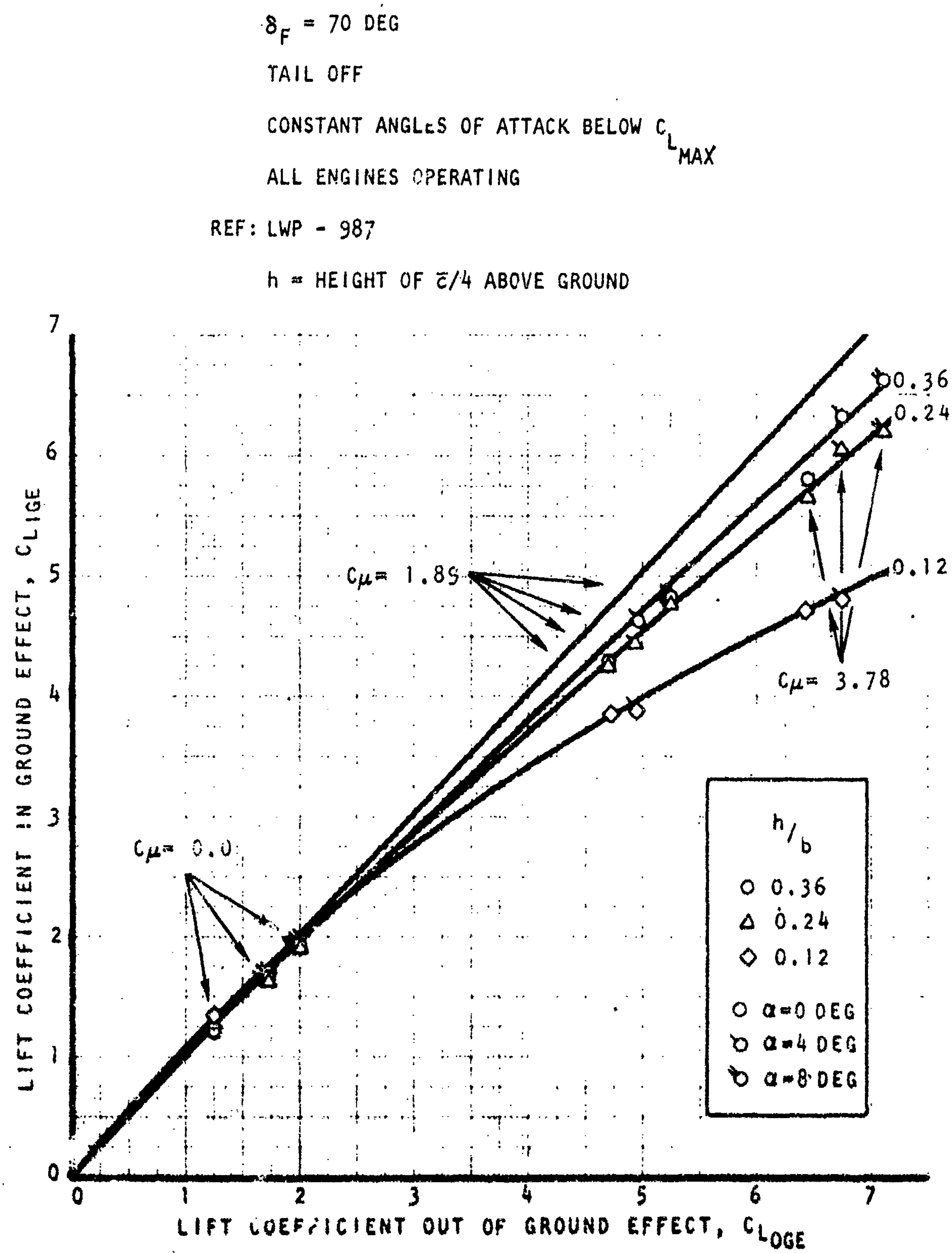


Figure 56. Ground Effect on Lift for Externally Blown Flaps

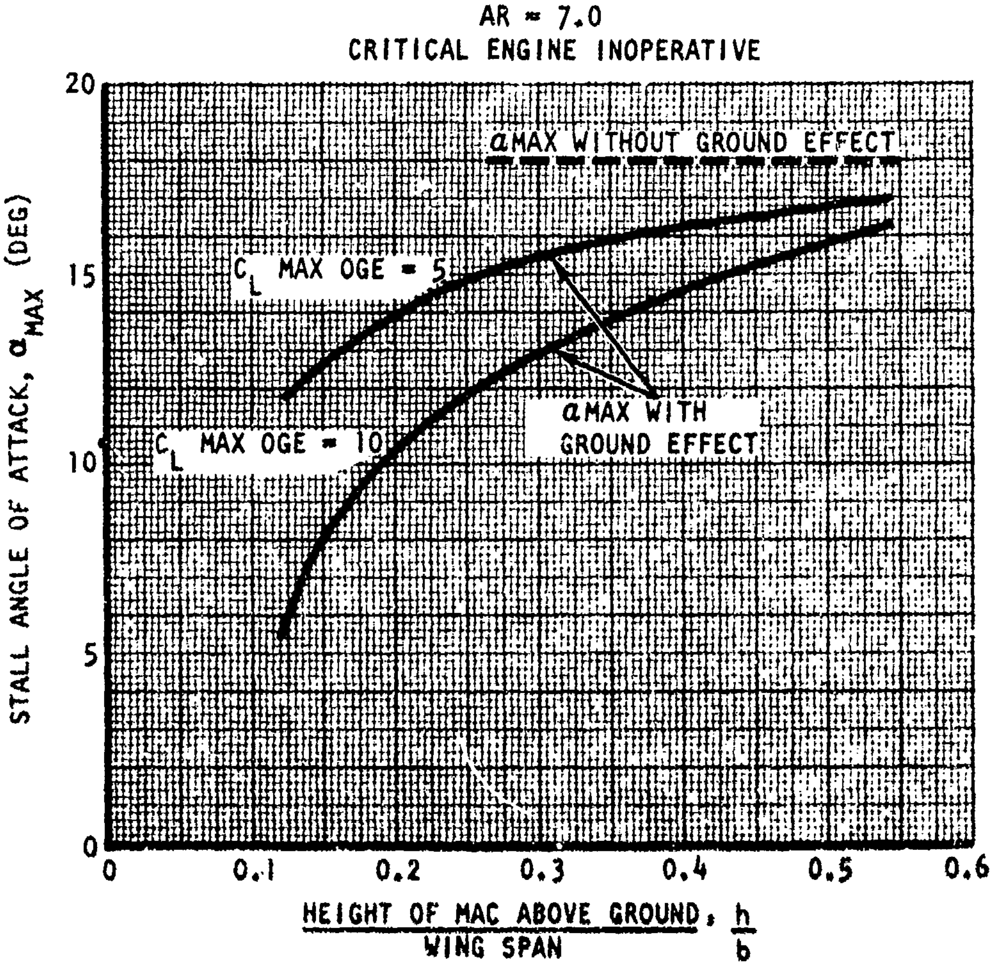


Figure 57. Ground Effect on Stall Angle of Attack

NO AILERON BLC

$\delta_F = 25^\circ/50^\circ$

$\boxed{C_{\mu_{PE}} = 0.50}$

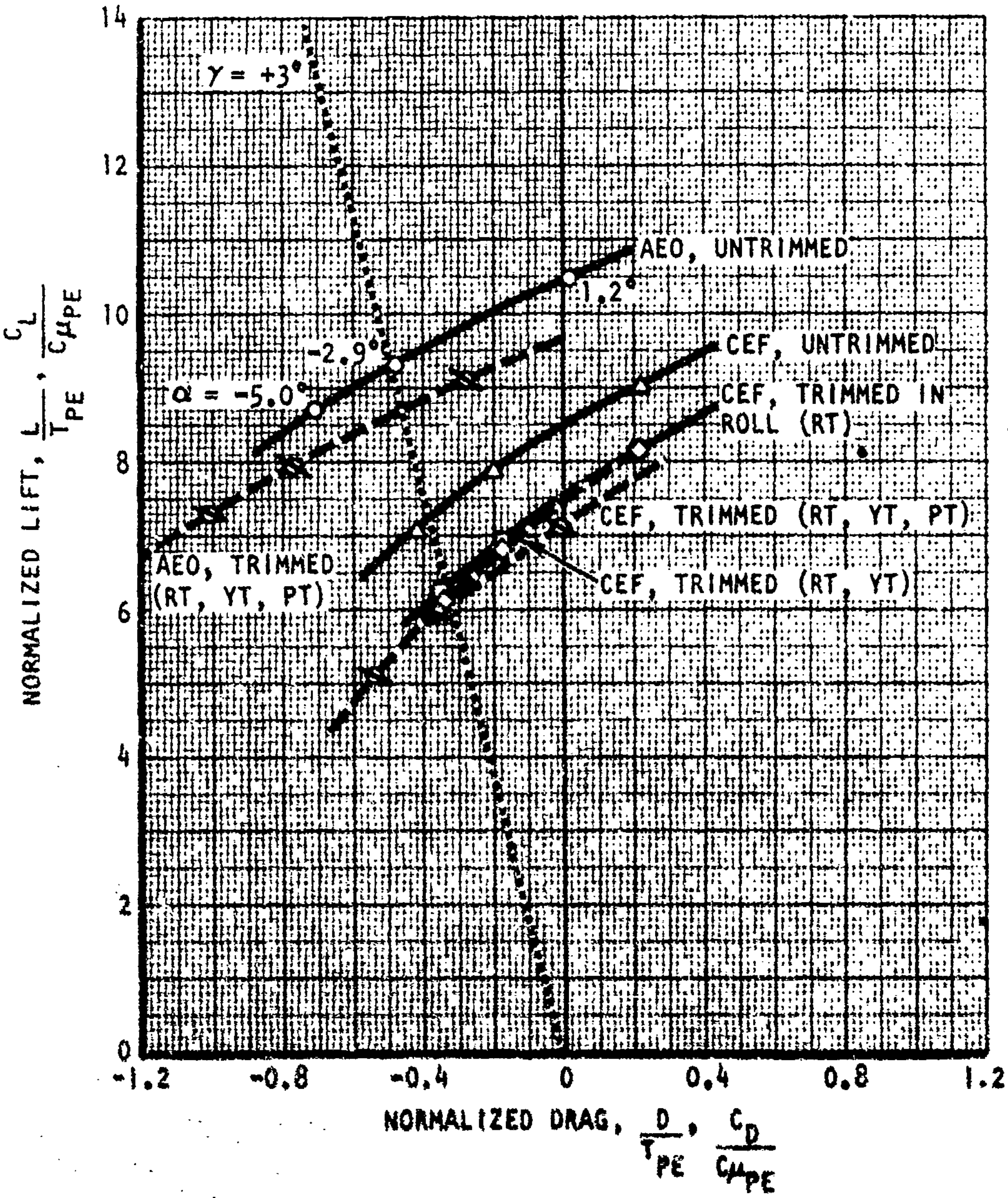
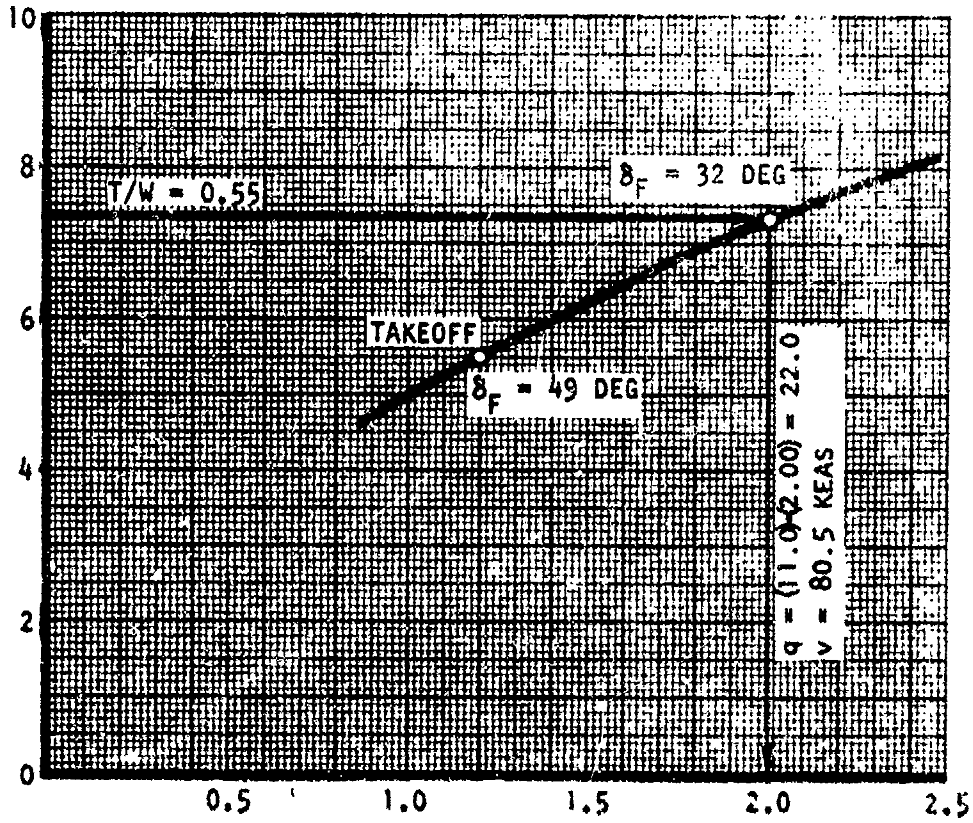


Figure 58. Full-Span Double-Slotted Flaps



$$\frac{C_L}{C\mu_{PE}} = \frac{L}{T_{PE}} = \frac{W}{T/4} = \frac{4}{T/W}$$

$$\frac{1}{C\mu_{PE}} = \frac{q}{T_{PE/S}} = \frac{q}{\frac{T_{PE}}{W}\ \frac{W}{S}} = \frac{q}{\left(\frac{\frac{T}{W}\ \frac{W}{S}}{4}\right)},\ \frac{q}{11.0}\ \text{for}\ \frac{T}{W} = .55,\ \frac{W}{S} = 80$$

Figure 59. Determination of Takeoff Speed

indicated. Takeoff speeds were then determined from this relation for given T/W ratios and W/S values as illustrated again in figure 59 with a sample design condition.

Landing flap angles are analyzed similarly and an example is shown in figure 60. The front flap segment is set at 25 degrees and at 35 degrees for comparison purposes, and the rear segment is varied over a large range. Flight safety margins for the landing case are applied and are different from those for takeoff. Also the landing lift and drag equilibrium along a descent flight path of γ = -4 degrees is indicated. This flight path represents very nearly a sink speed of 10 feet per second. Only conditions with the critical engine failed are considered.

Similar to the takeoff case, also in the landing case the intersection between the safety conditions and the descent flight path condition was used to determine the optimum lifting capability that satisfies all constraints. It is seen that no performance benefit is obtained in going from a 25 to 35 degree setting of the first flap segment.

The optimum lifting capability in the landing approach is obtained from such plots and is presented in figure 61. In this figure the approach condition is also compared to the takeoff condition. It is seen that the approach speed is lower than the takeoff speed at the same aircraft weight and same engine power setting in the engine-out operation. In order to preserve waveoff capability, however, the approach speed may not be less than 90 percent of the takeoff speed as is described in volume III. In that volume, additional details regarding takeoff and landing are discussed.

6.2 STABILITY AND CONTROL

6.2.1 LATERAL CONTROL

The lateral control system consists of full-span spoilers and insert ailerons in the second flap segment of the double-slotted flap at the wingtip. The inboard spoilers are also used for direct lift control (DLC), and an interconnect between roll control and DLC exists to minimize the lift loss due to roll control for flap deflections used during approach and landing. This interconnect does not exist for takeoff flap settings or for retracted flaps. The insert ailerons provide the initial roll control when the flaps are retracted, their motion is not phased with flap deflection. No BLC is used in the roll control system on the basis of a study reported in the supplement to volume I.

The sequence of the spoiler deflection for roll control is such that the spoiler at the wingtip is moved first; thereafter, the spoiler panel just inboard of it, and so on, so that the lift loss due to spoiler actuation is

NO BLC

DOUBLE SLOTTED FLAPS

$C_{\mu_{PE}} = 0.50$

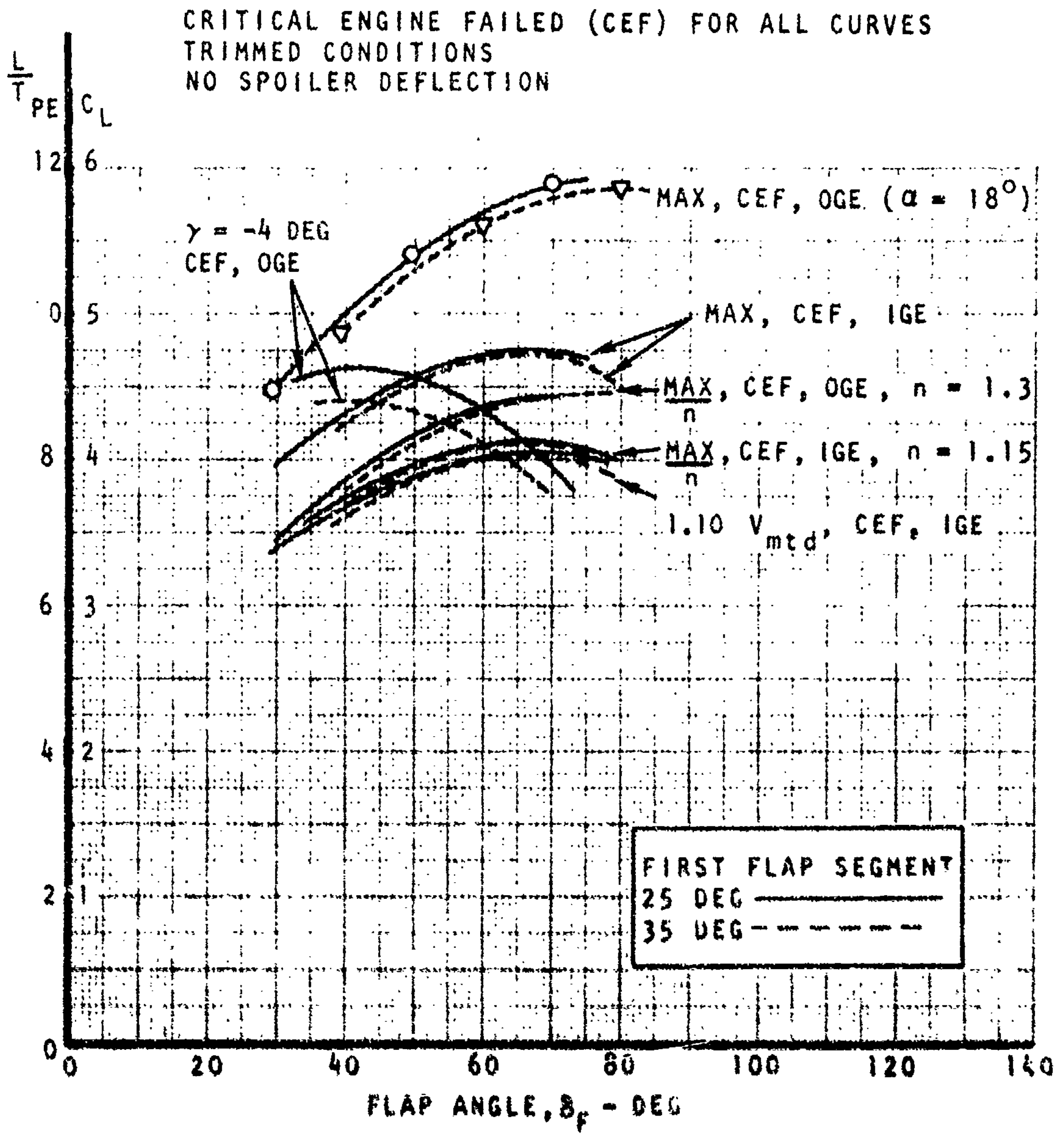


Figure 60. Determination of Landing Flap Angle

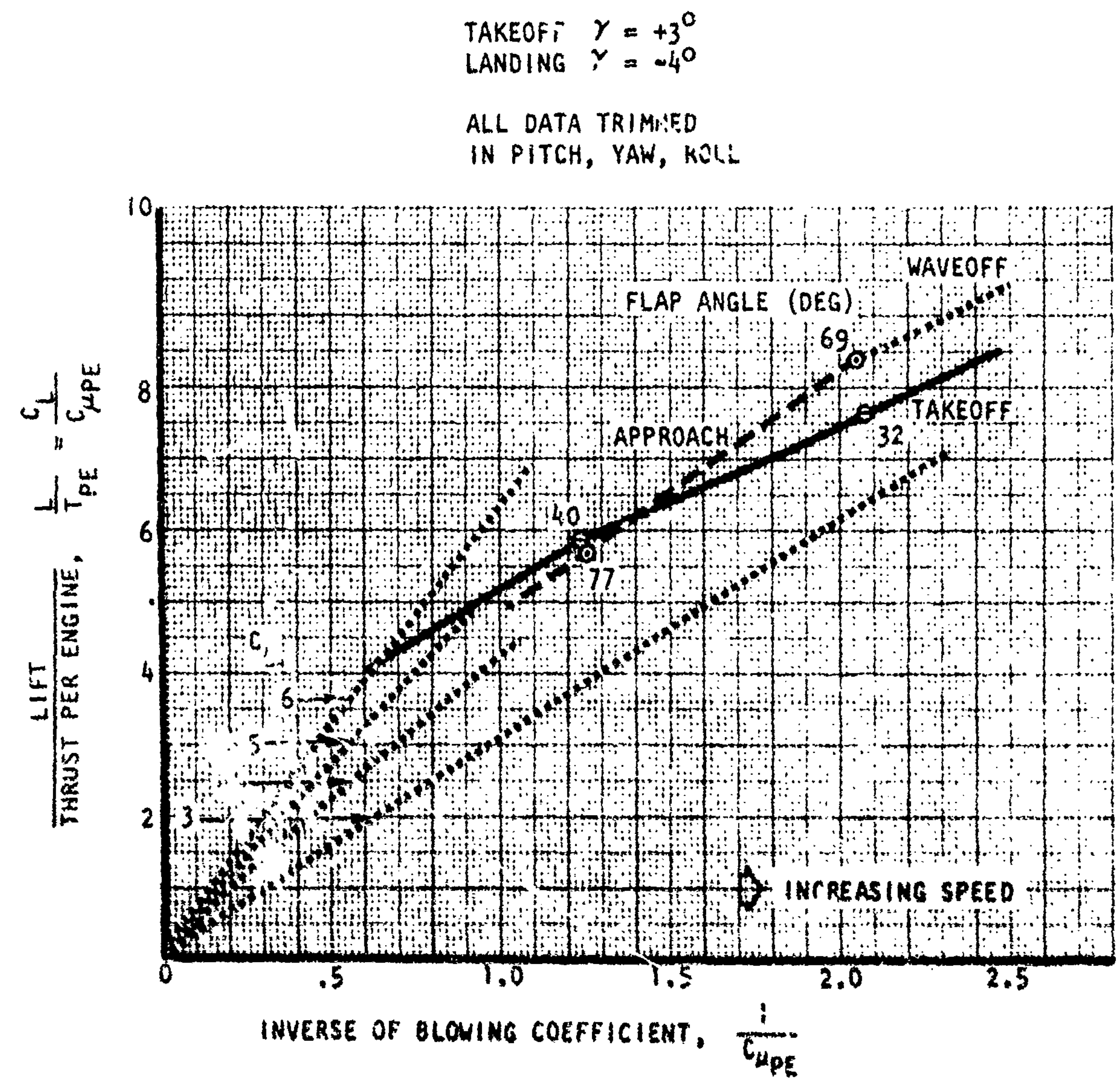


Figure 61. Lift vs. Inverse of Blowing Coefficient for Takeoff and Approach Speeds with the Critical Engine Failed

kept as small as possible for the majority of roll control inputs. In the flaps-up case, spoiler deflection is preceded by the deflection of insert ailerons to obtain the required initial response.

The lateral trim system in the flaps-down case consists of a change in the angular setting of the last segment of the double-slotted flap at a portion of the flap span. This portion is the second of the four spanwise flap panels, counting from inboard to outboard, to minimize the induced yawing moments. Large rolling moments need to be trimmed out in case of a failed engine.

The most stringent roll control requirements in the STOL mode for a class II Medium Weight Transport are roll acceleration requirements. The available roll acceleration for the aircraft is compared with requirements from various different sources in figure 62 for normal operation. It is seen that the roll acceleration requirement from NASA TND-5594 (reference 11) is met for all aircraft weights, and that the requirements from MIL-F-83300 (reference 12) are met for weights above about 112,000 pounds. The comparisons are made at the minimum approach speeds consistent with each of these weights. Increasing the approach speed above this minimum increase the available roll acceleration, and in that case all requirements are met.

The available roll acceleration is determined from

$$\ddot{\phi} = \frac{\mathscr{L}}{I} = \frac{C_\ell \, q\, Sb}{I} = \frac{C_\ell}{C_{L_{trim}}} \frac{bW}{I}$$

where $C_\ell/C_{L_{trim}}$ is taken from figure 63 which, in turn, is substantiated in the appendix to volume I.

The roll control requirements are shown in terms of the acceleration without damping, such as would exist in a theoretical step input at time zero. These values are all computed from various bank angle requirements to be met after various time lapses, including the effect of damping. The computation is made to obtain a comparison of requirements on a consistent basis. Herein it is assumed that the spoilers start moving 0.10 second after pilot control initiation, and that the desired roll control is obtained 0.40 second after the control initiation. A roll time constant of $\tau_R = 0.70$ is used which is typical for the landing configuration. The roll control requirements from AGARD 408A (reference 13) are those for "normal operation," and those of NASA TND 5594 are called out as "satisfactory operation" in that document. The latter reference quotes a bank angle for a "satisfactory operation level" with a wheel deflection of 60 degrees as compared to the maximum available wheel deflection of 90 degrees stipulated in that reference. In figure 62 the required acceleration from this reference was increased by 90/60 to represent a maximum control input consistent with 90 degree wheel deflection.

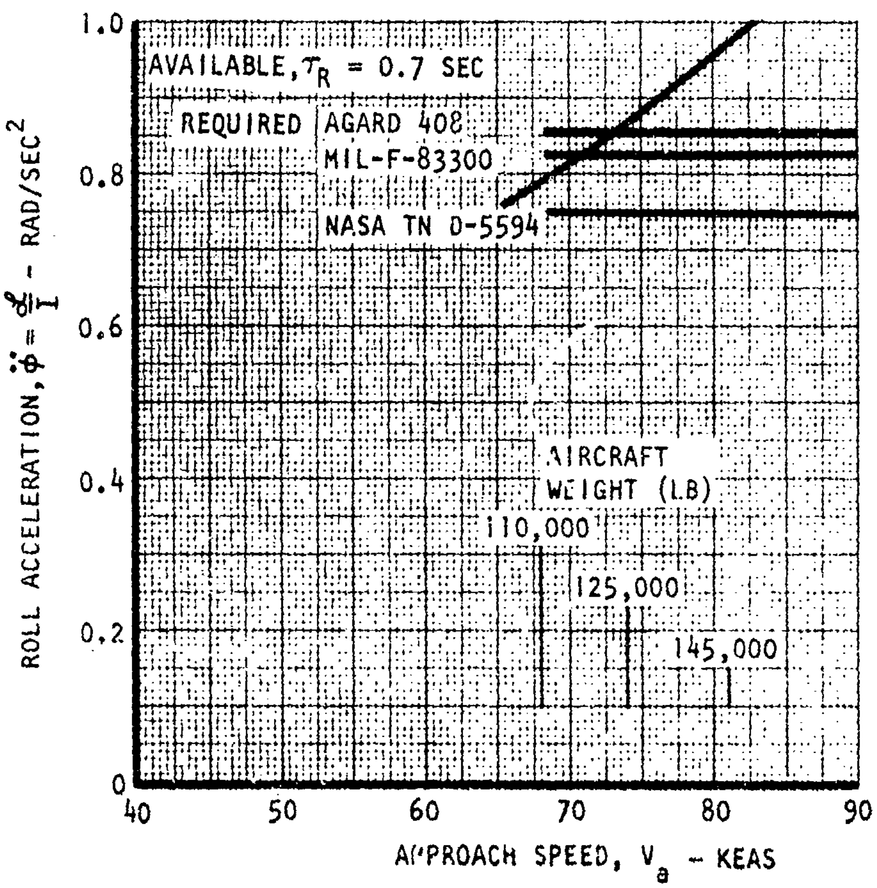


Figure 62. Roll Acceleration Characteristics in STOL Mode
(Normal Operation)

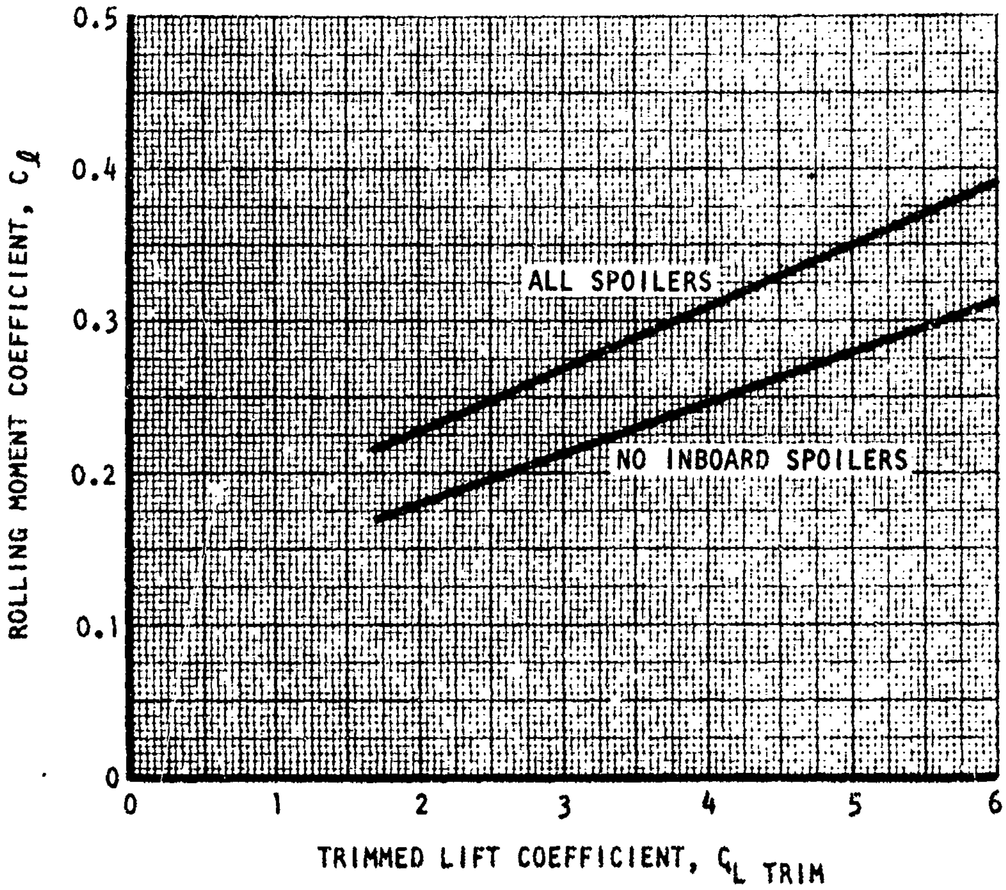


Figure 63. Available Rolling Moment Coefficient in the STOL Mode

It is seen that under these circumstances, the roll control acceleration requirements from the three different sources are very similar in magnitude.

The roll acceleration capability after engine failure but before lateral trimming with asymmetric flap is shown in figure 64. The capability exceeds the required level of MIL-F-83300. Bank angle response after engine failure and pilot corrective action is shown in figure 65. Herein, pilot action is initiated approximately 1 second after engine failure, and full roll control is obtained in approximately 1.4 seconds after failure. Engine transients used are those pertaining to an instantaneous fuel loss. After trimming of the engine failure rolling moment with asymmetric flap, the available roll acceleration exceeds the capability shown in figure 64.

Additional details with regard to the engine failure transients are given in volume III. Engine failure moments used in the analyses are shown in figure 66 as a function of the power effect on lift, ΔC_{L_P}. The effect of rudder deflection due to trimming in yaw is included in the rolling moment. The rolling moment is also expressed in the total aircraft lift coefficient in figure 67 and compared therein with the available roll control. The difference between the available rolling moment and the engine failure moment is used to compute the available roll acceleration.

6.2.2 LONGITUDINAL STABILITY AND CONTROL

The pitch control system incorporates a double-hinged elevator, with the first hinge at the 55-percent chord station of the stabilizer, and the second hinge at the 75-percent chord station. The maximum deflection of the rear element is 25 degrees with respect to the first element, and the maximum deflection of the first is 25 degrees with respect to the stabilizer. At low-speeds, both elements deflect in a one-to-one ratio to obtain adequate control power. At high speeds the first hinge is locked to prevent the aircraft from being sensitive to pitch control.

The stabilizer is movable for trimming purposes and to obtain the proper setting to reach the maximum downward lift capability during STOL landing approaches. The rate of change of the stabilizer is slow. The stabilizer leading edge is deflected upward when the wing flaps are extended in order to increase the maximum download on the tail.

The aircraft incorporates a DLC system for landing touchdown control when the flaps are deflected for landing, using large span spoilers. An interconnect between DLC and roll control deflection is provided to minimize lift loss due to roll control. The DLC system is operated by a thumb controller at the power levers. No DLC control is available when the flaps are set in the takeoff position so that a high lift-to-drag ratio is available during climb and waveoff.

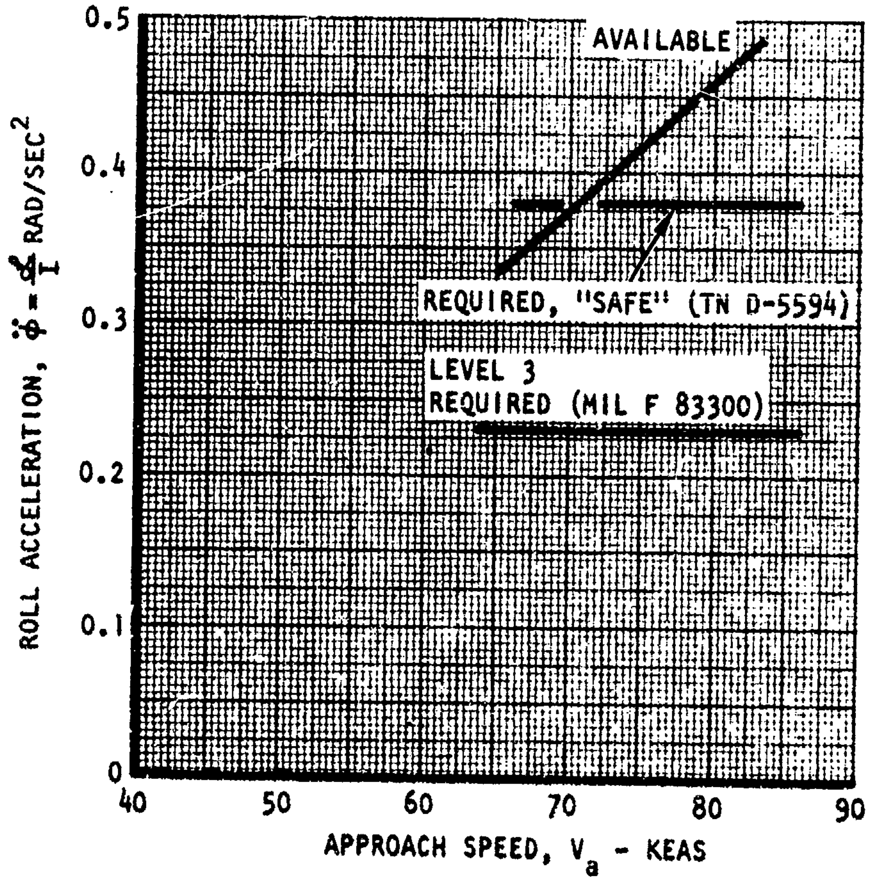


Figure 64. Roll Acceleration Characteristics in STOL Mode
(Critical Engine Failed)

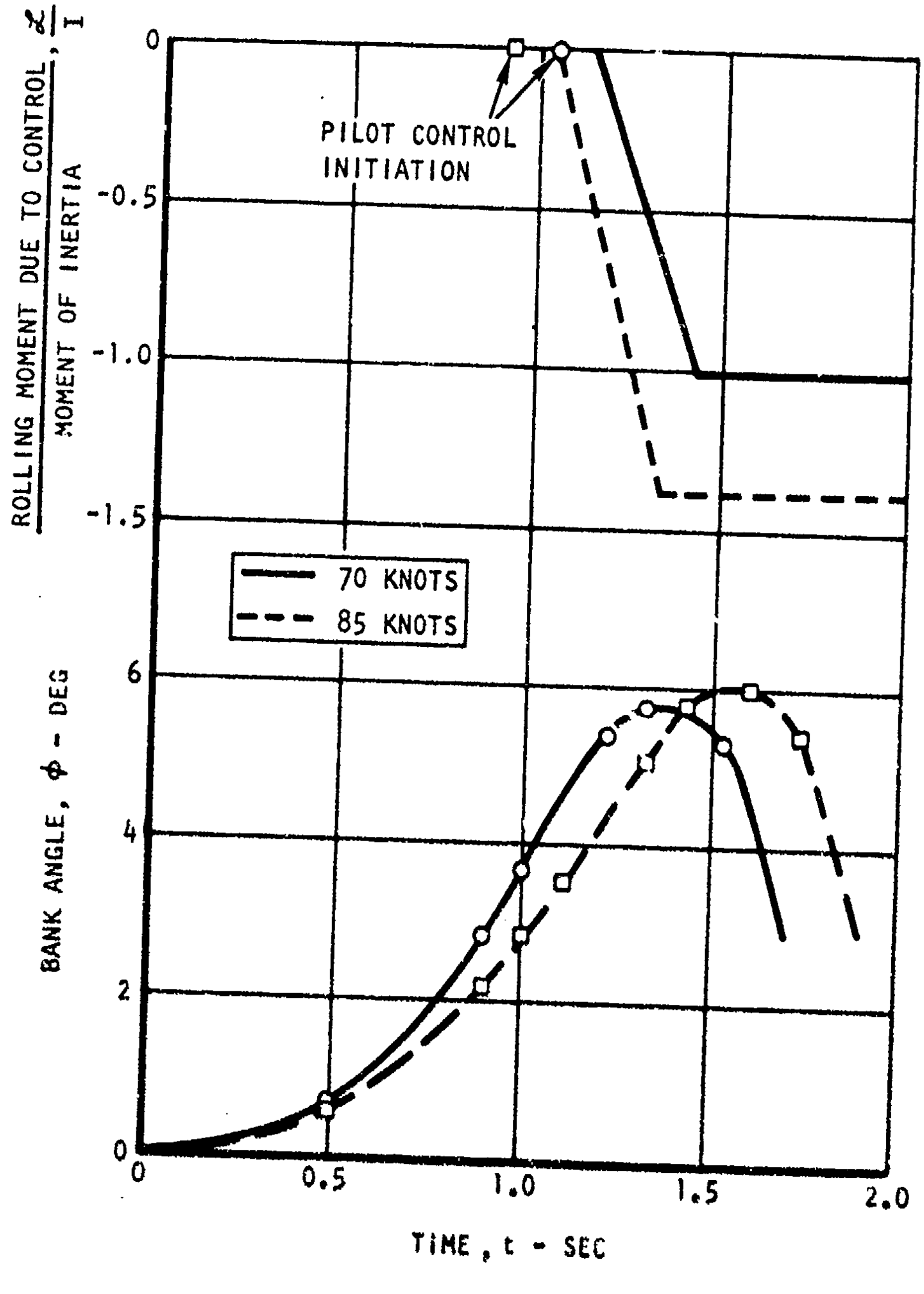


Figure 65. Bank Angle Response After Engine Failure

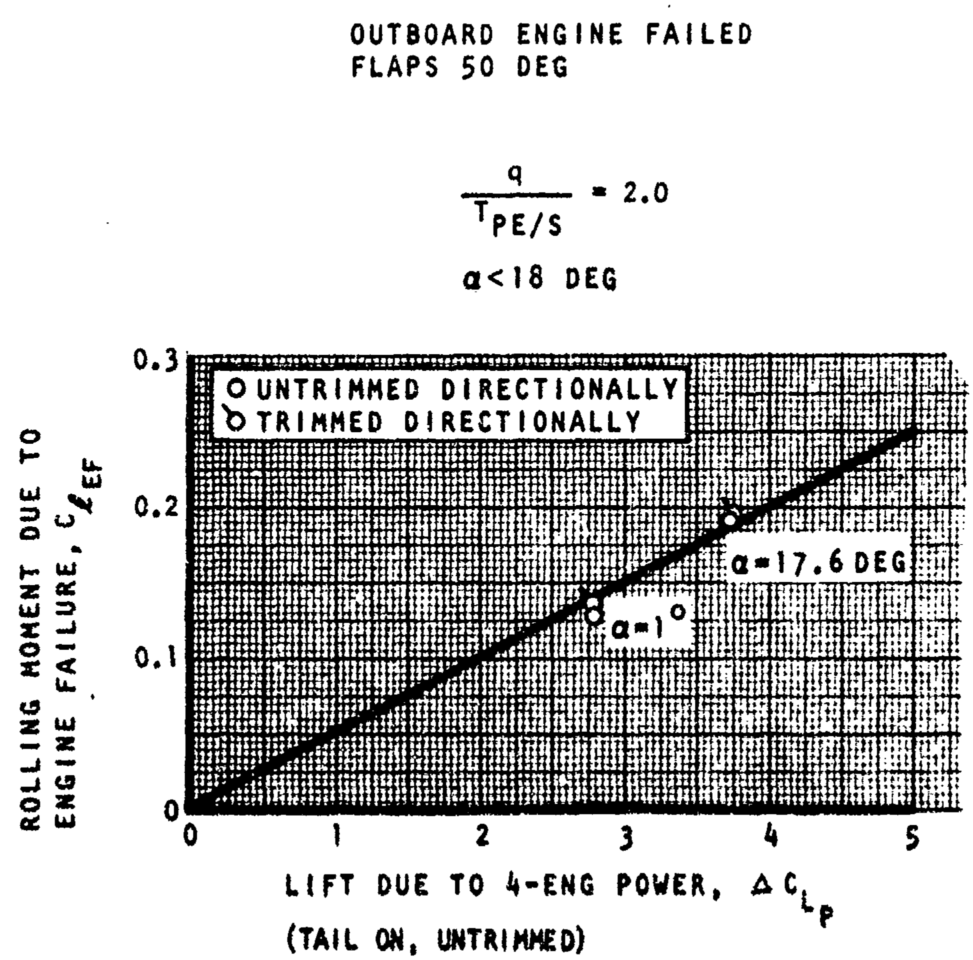


Figure 66. Rolling Moment Due to Engine Failure With and Without Directional Trimming

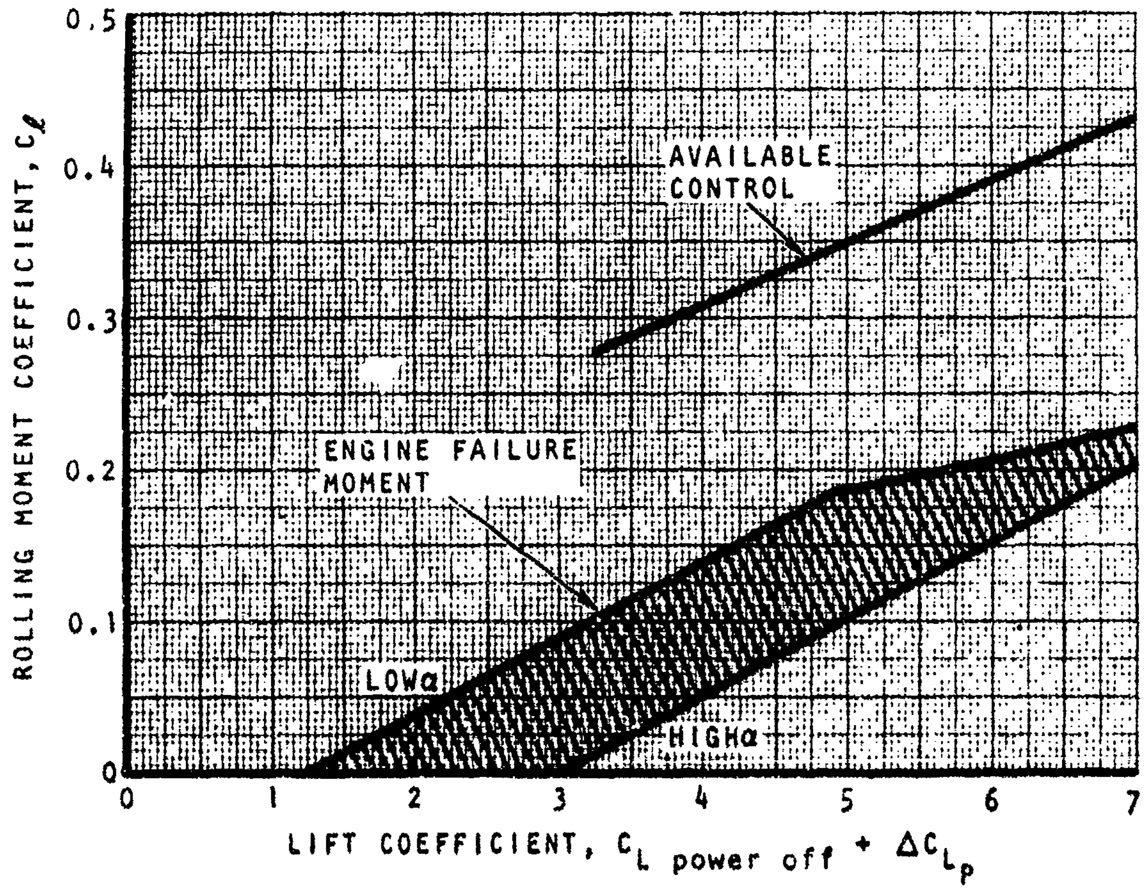


Figure 67. Rolling Moment Characteristics After Engine Failure (Untrimmed)

A direct drag control (DDC) system is provided and operated by a thumb controller at the control column. This controller operates the deflection of the aft segment of the double-slotted flap. The rate of decrease of the angle of this segment is in the order of 12 degrees per second so that the setting can be changed from the landing position to the waveoff position in approximately 3 seconds. This provides adequate forward acceleration during waveoff with the critical engine failed. During retraction, the flap loads aid in reducing the flap angle. The rate of increase of the flap angle is considerably slower but satisfactory for use in automatic glide path control during landing approaches where the aircraft is generally operated at the back side of the power curve. The setting of the first flap segment is not affected by DDC.

A summary plot of stability and control criteria in terms of weight and CG locations is presented in figure 68. The figure shows limits of neutral stability, nosewheel liftoff, pitch acceleration, and conditions where maximum lift can be reached in the STOL mode. The center-of-gravity locations of the aircraft are compared with these limits. It is seen that the maximum lift can be reached throughout the weight range of the aircraft, and that the aircraft is stable in all cases. However, at low weights the desired pitch acceleration (approximately $M/I = 0.30$ rad/sec^2) and nosewheel liftoff is not attained at the low speeds that are associated with the low weights. The speeds pertain to takeoff or approach speeds as determined from the takeoff and landing ground rules such as speed margins, maneuver margins, and takeoff or waveoff climb capability, and are therefore low at low weight. This, in turn, results in a low dynamic pressure and reduction in pitch control power. If the speed is increased at these low weights above the level required for margins and climb, adequate pitch control is available to meet both the desired pitch acceleration and the nosewheel liftoff criteria. This increase of speed does not increase the takeoff and landing field length beyond the length required for the heavy weights in the STOL mode, and therefore this increase is considered to be acceptable. In the subsequent portion of this section, the various limits are described in more detail, and additional basic information is given.

Nosewheel liftoff characteristics are presented in figure 69. The desired rotation speed is compared with the speed at which the nose gear can be lifted as a function of aircraft weight and CG location. The desired rotation speed is chosen to be 95 percent of the takeoff speed in order to assure a proper liftoff angle of attack at the moment the takeoff speed is reached.

The nosewheel liftoff capability is computed by using the maximum download at the tail. The maximum lift coefficient in the downward direction is $C_{L_{max}} = 1.935$ as shown by the tail aerodynamic characteristics presented in figure 70. The tail incorporates a plain leading edge flap hinged at the 15-percent chord station.

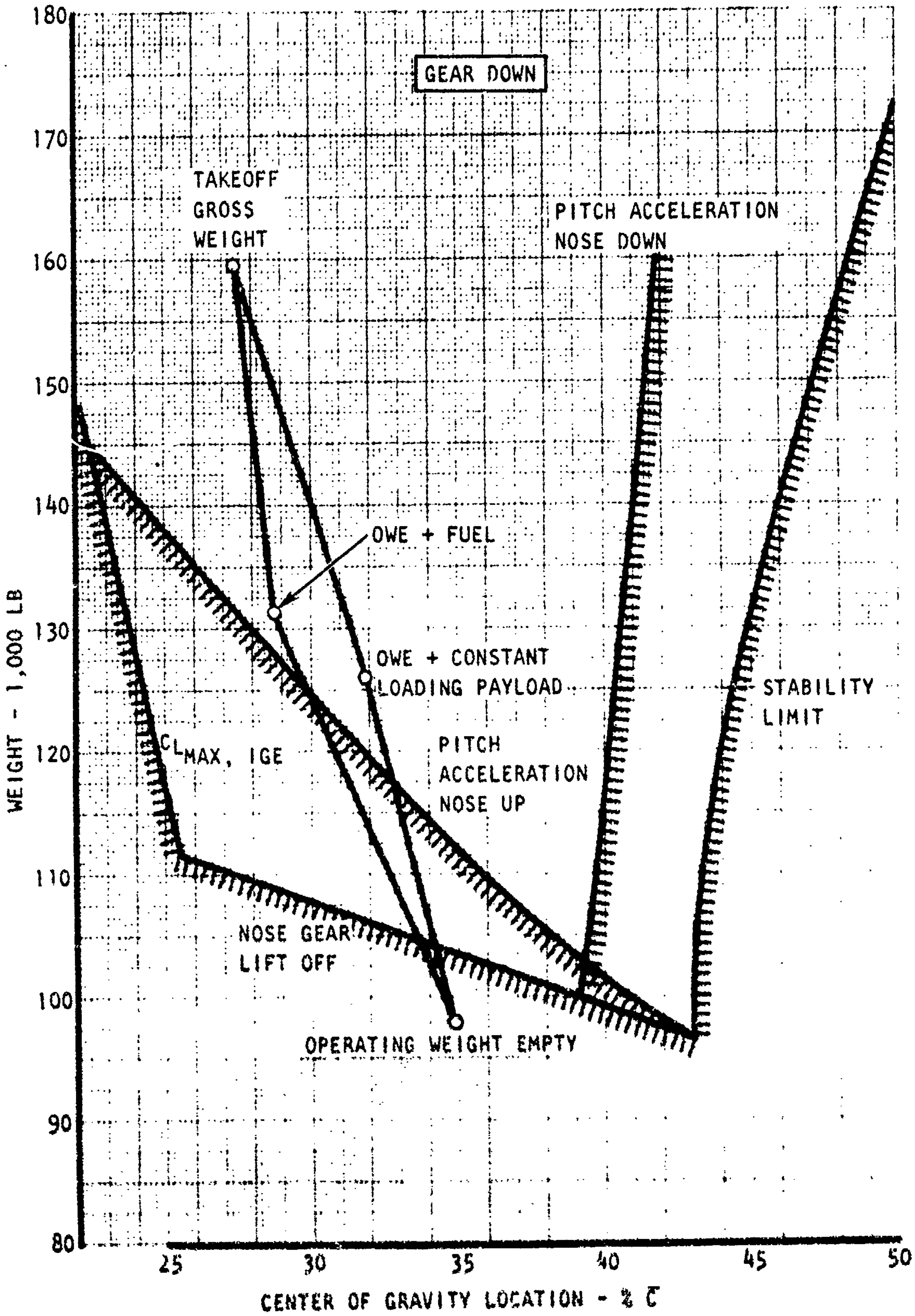


Figure 68. Center of Gravity Limits

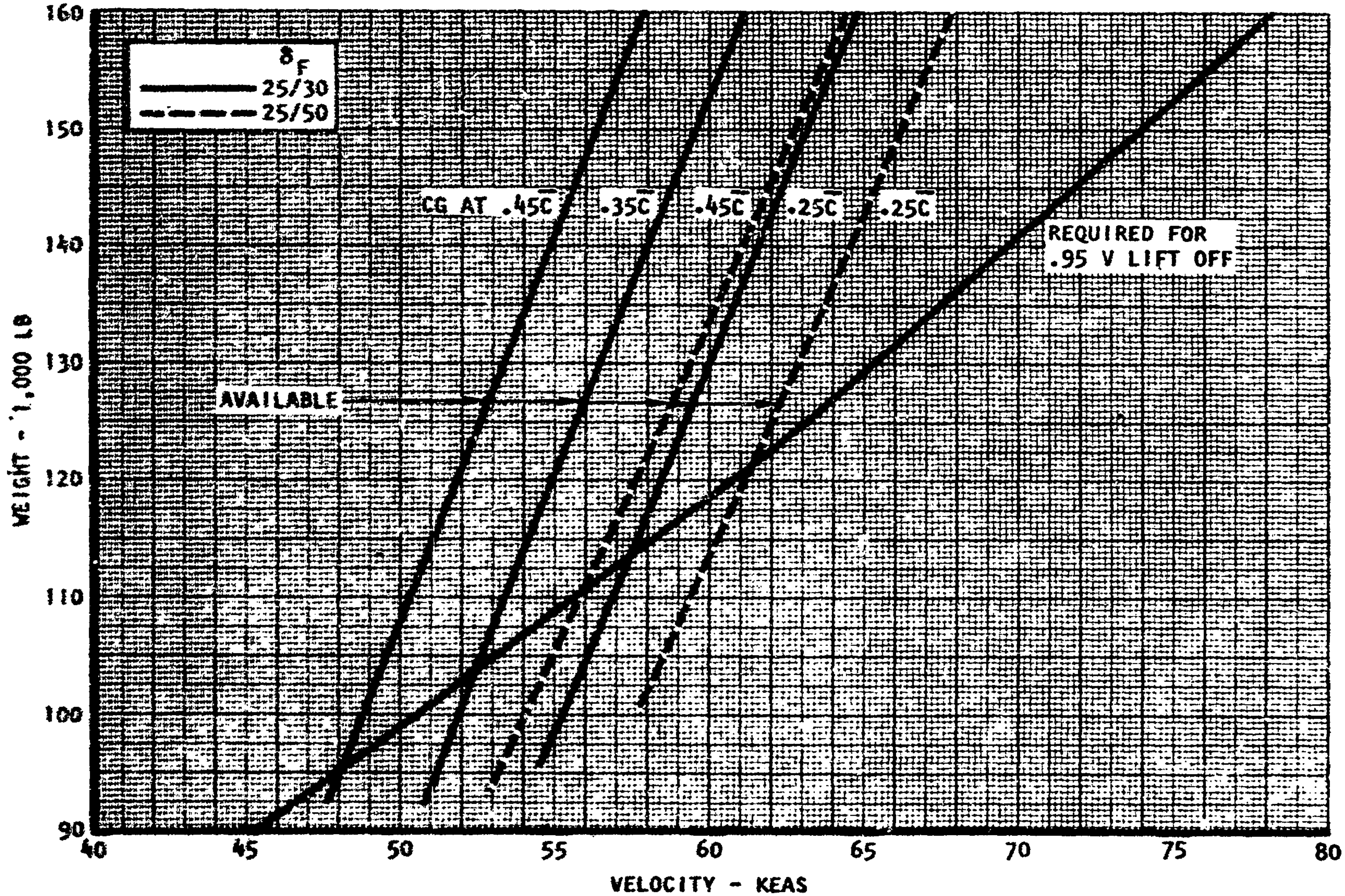


Figure 69. Required and Available Control for Nose Wheel Liftcff

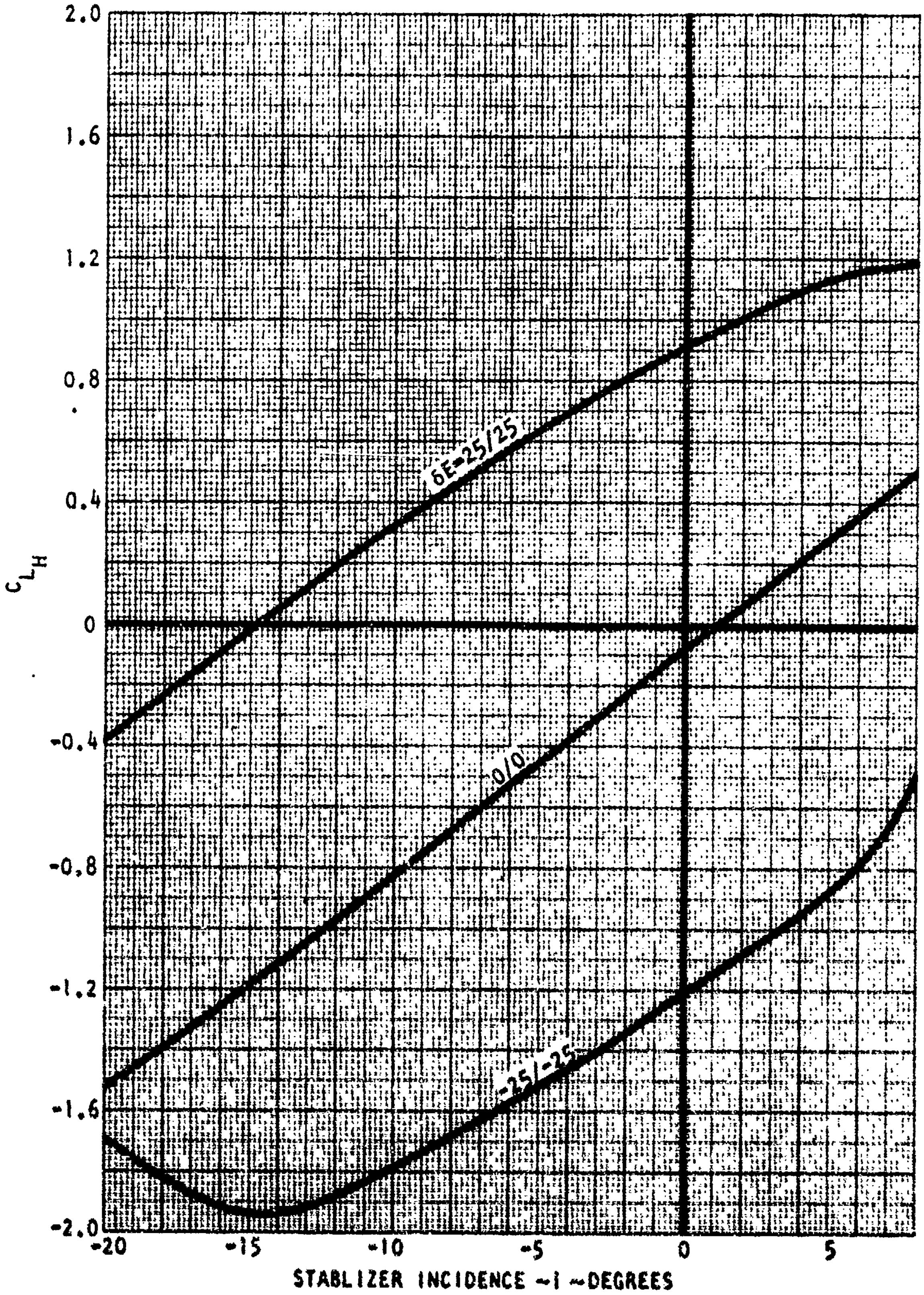


Figure 70. Horizontal Tail Lift Characteristics

Noseup and nosedown pitch acceleration capability, presented in figure 71, are compared with the desired level of figure 37 of NASA TND-5594. It is seen that the noseup acceleration is the more critical of the two, and the desired angular acceleration is met at the higher of the speeds shown. To obtain the desired level at low weights, the approach speed should be higher than that which meets speed and maneuver safety margins and waveoff capability. A reduction in approach speed, due to this criterion, may be possible by the substitution of normal for angular acceleration through power or direct-lift control means. The NASA reference implies a restriction of the desired angular acceleration level for this weight-size class vehicle to maintain passenger comfort at aft stations. Flight simulator investigations of the impact of DLC on the required pitch acceleration level and justification of the level is indicated. The speeds, flap angles, maximum download stabilizer settings, and moments of inertia used are presented in figure 72 as a function of aircraft weight. Tail-off characteristics and downwash angles used, shown figures 73 and 74, pertain to an approach power setting.

The elevator angle required to reach the maximum wing lift is presented in figure 75. The $C_{L_{max}}$ in ground effect with full four engine power is presented in figure 76 and is associated with angles of attack presented in figure 77. Maximum four-engine power is used in case maximum power is applied just before touchdown. The stabilizer setting used is the same as that for pitch acceleration during normal landing.

The $C_{L_{max}}$ in ground effect can probably be demonstrated in flight by obtaining the minimum speed with the critical engine failed and the airplane flying as close as possible to the ground with the gear just off the runway surface. With the gear in this unloaded condition, a ground angle of 15 degrees is available which is adequate for the demonstration. A ground angle of 10 degrees is available with the gear in the design load condition which is adequate during descent. In that case, the maximum angle of attack can still be obtained up to approximately 15 degrees, as shown in figure 78.

Aircraft neutral point locations for approach conditions are presented in figure 79. At lower weight, i.e., at lower flight speeds the blowing coefficient C_μ becomes larger which reduces the stability level.

6.2.3 DIRECTIONAL STABILITY AND CONTROL

The directional control capability is provided by a double-hinged rudder without BLC. The forward hinge is located at the 55-percent chord station of the vertical tail, and the second hinge at the 75-percent chord station. At low speeds, the first and second section of the rudder move in a 1:1 ratio; the maximum deflection of the second hinge is 25 degrees with respect to the first section of the rudder, and the maximum deflection of the first hinge is 25 degrees with respect to the vertical tail surface. At high speeds, the first hinge is locked to avoid excessive directional control sensitivity.

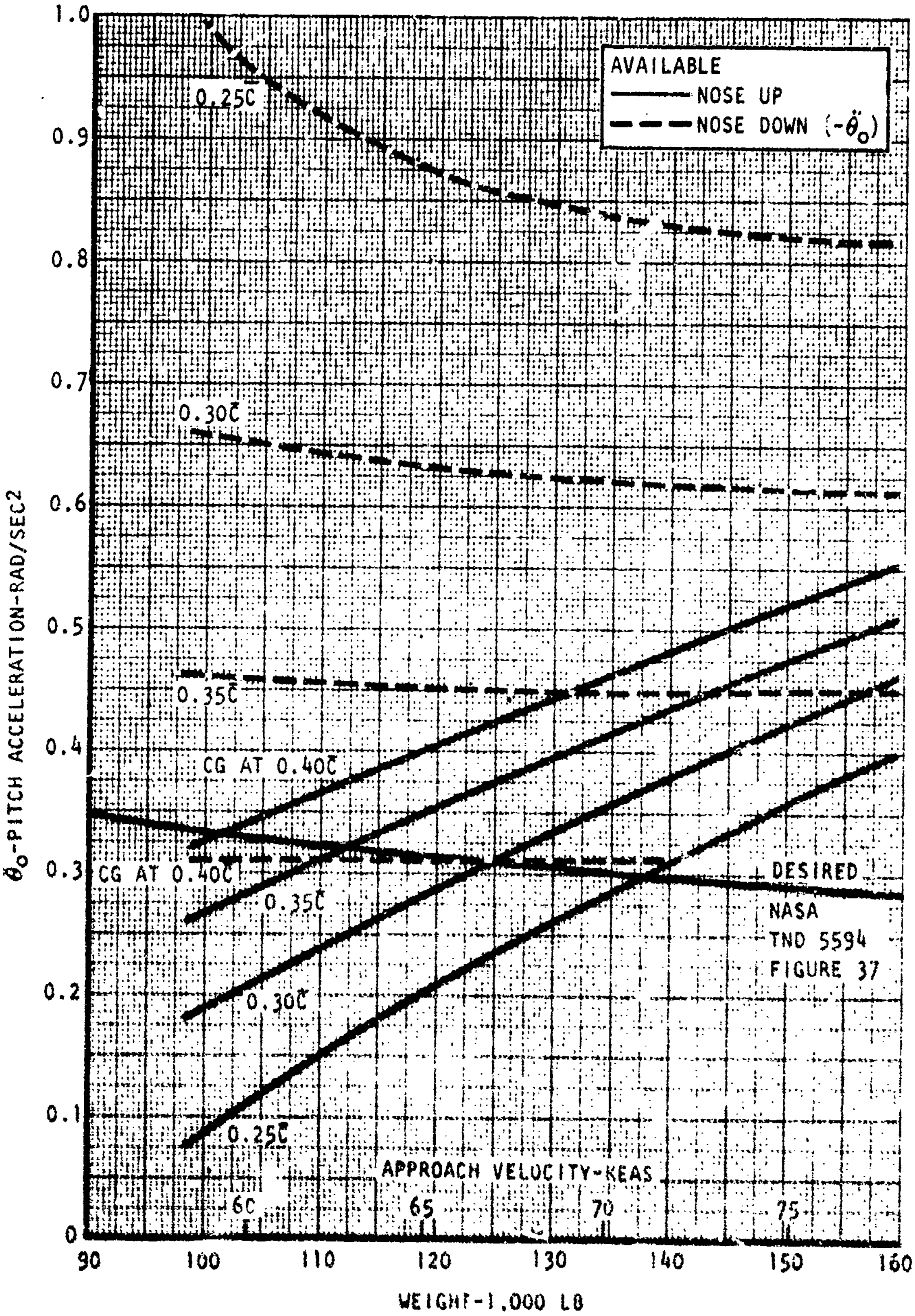


Figure 71. Pitch Acceleration Characteristics

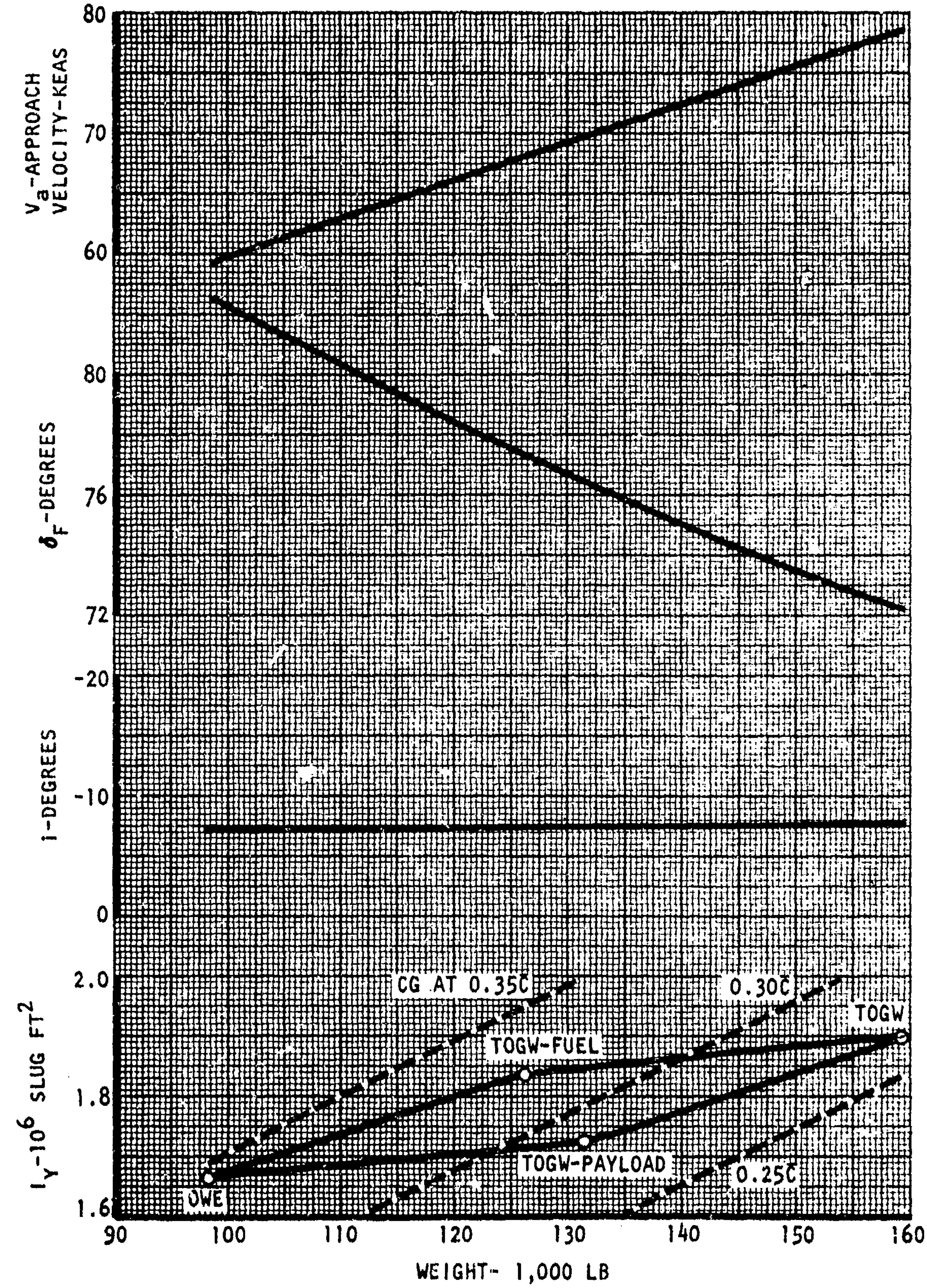


Figure 72. Approach Conditions and Moments of Inertia

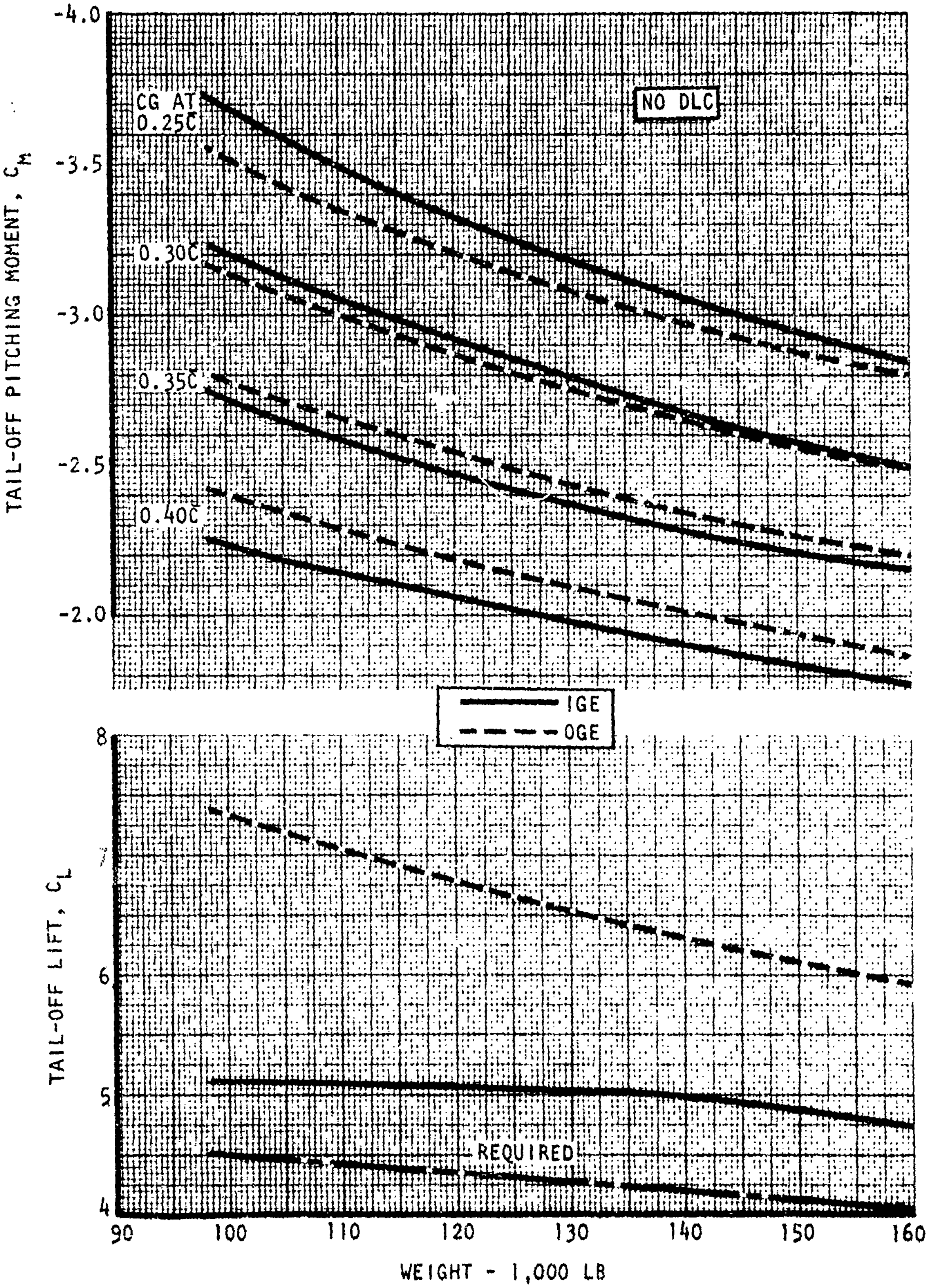


Figure 73. Aerodynamic Lift and Pitching Moment During Approach at α = 0

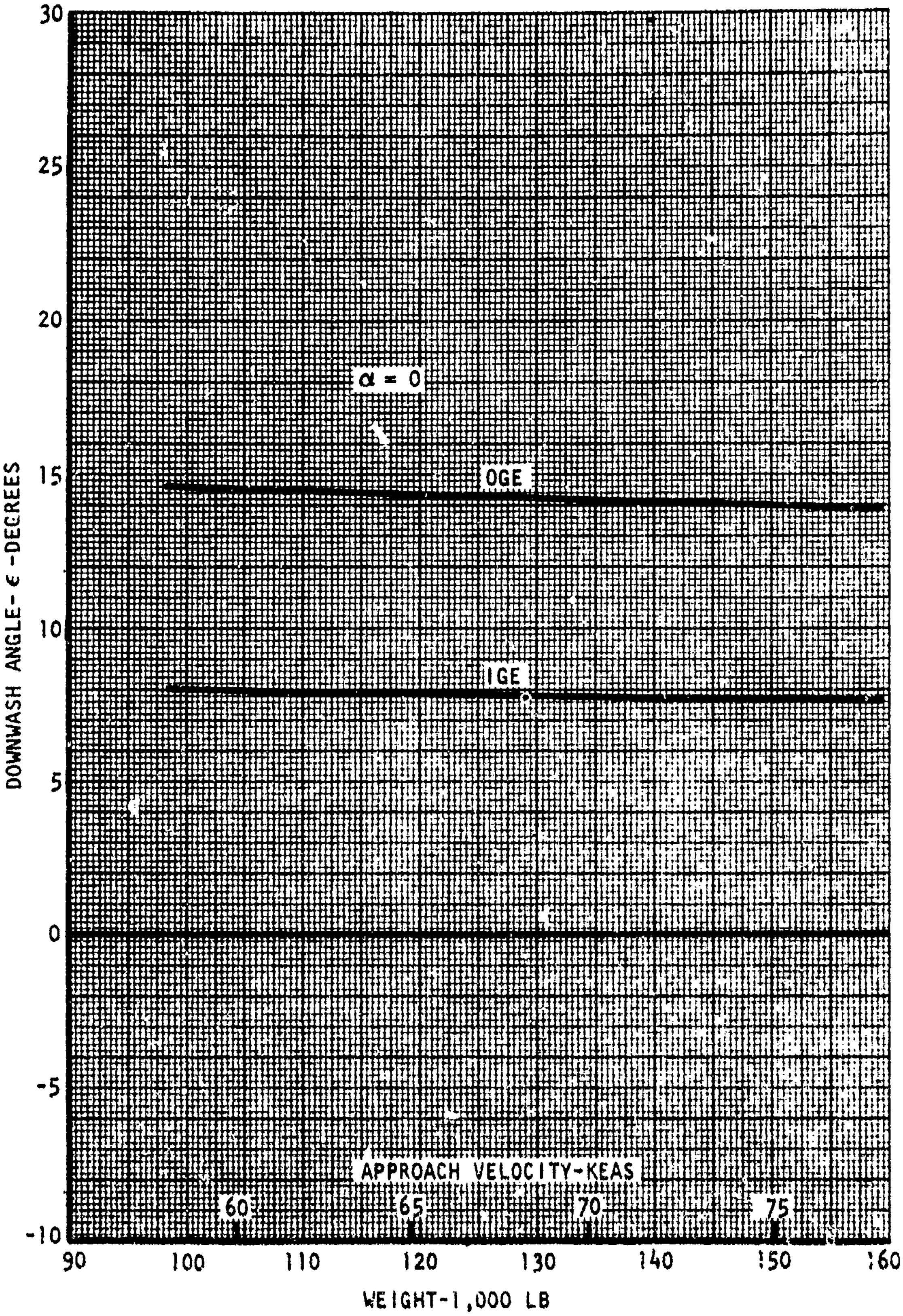


Figure 74. Downwash Characteristics

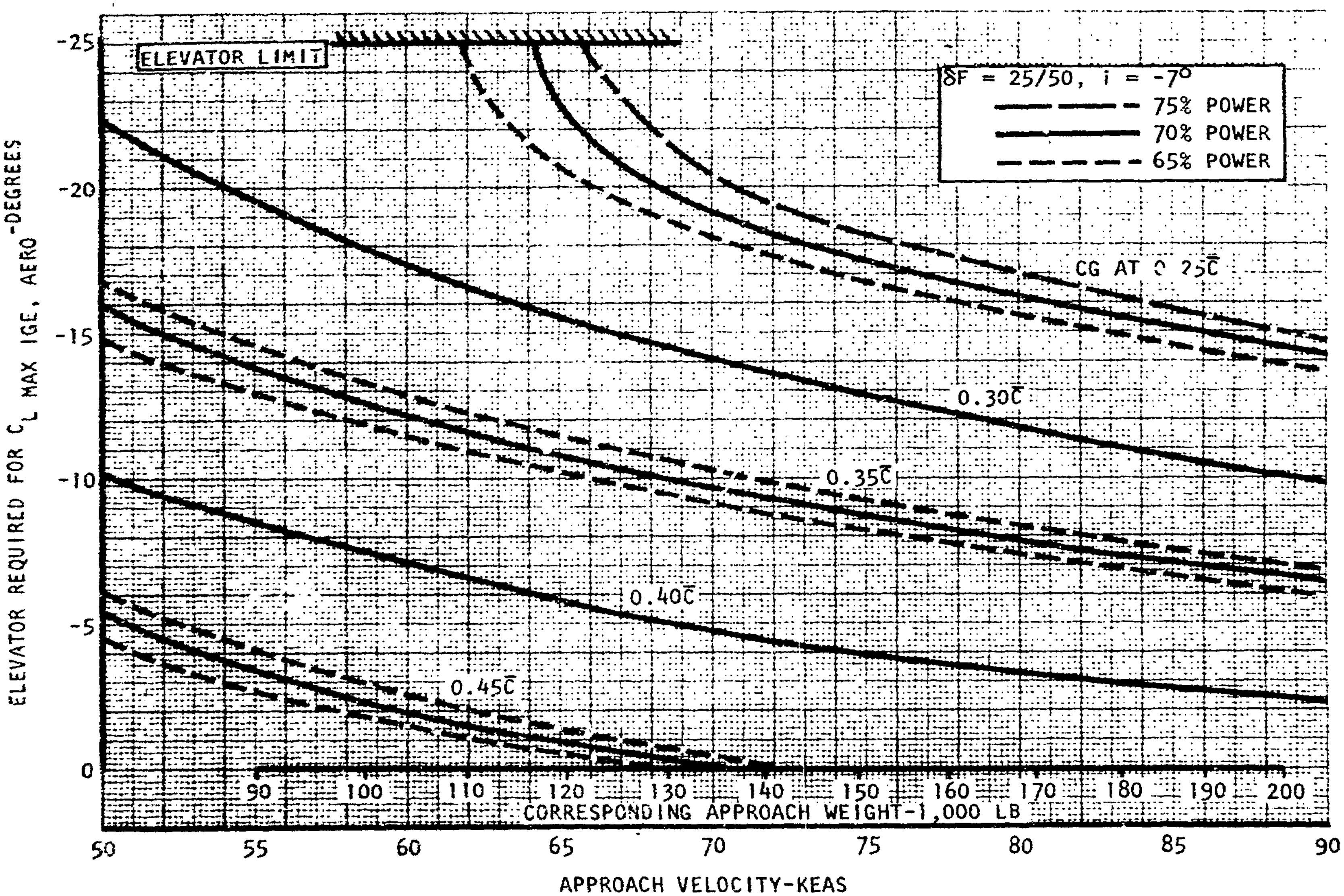


Figure 75. Elevator Angle Required to Reach Maximum Lift

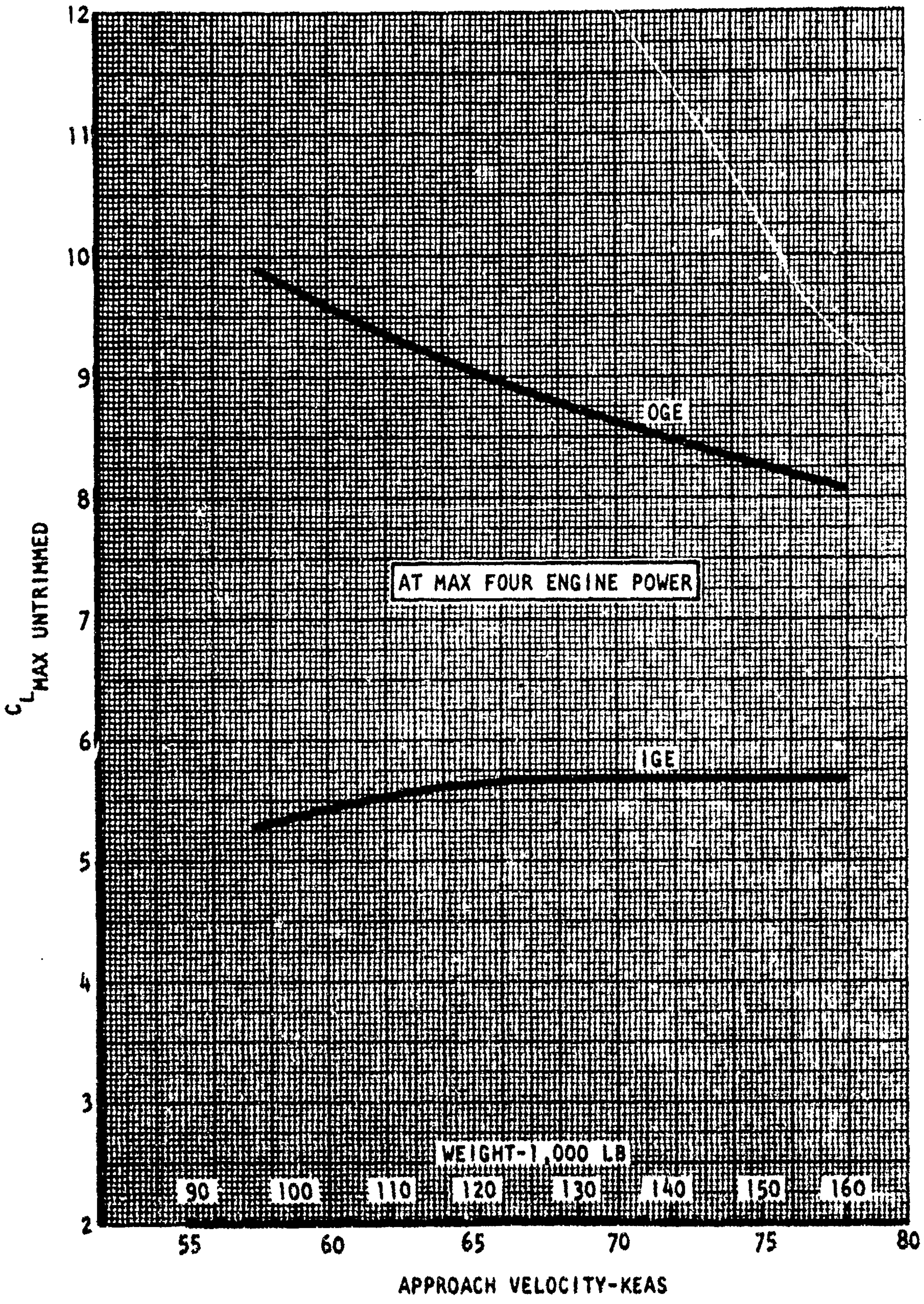


Figure 76. Maximum Four Engine Lift

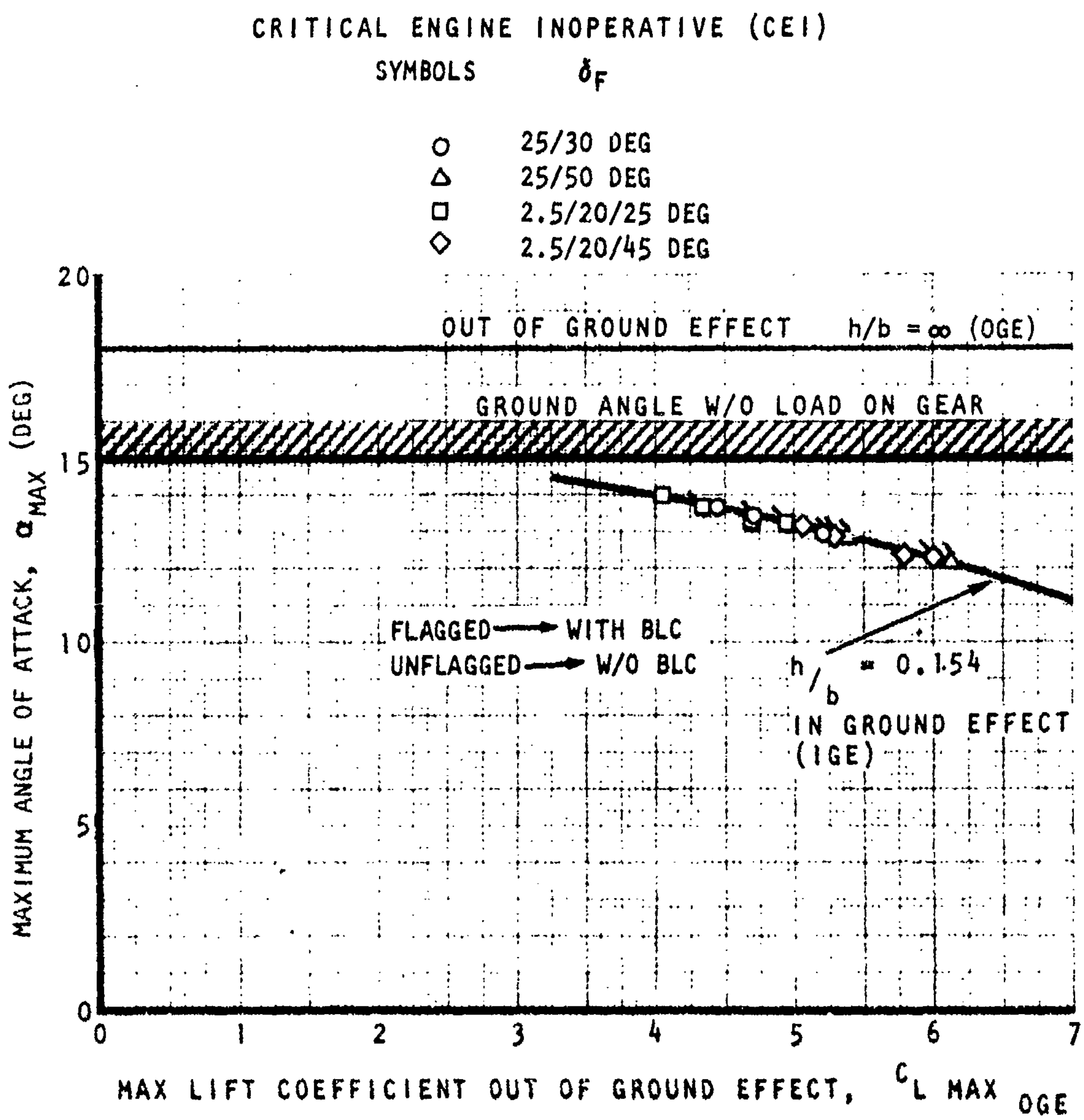


Figure 77. Effect of Ground Proximity on α_{max}

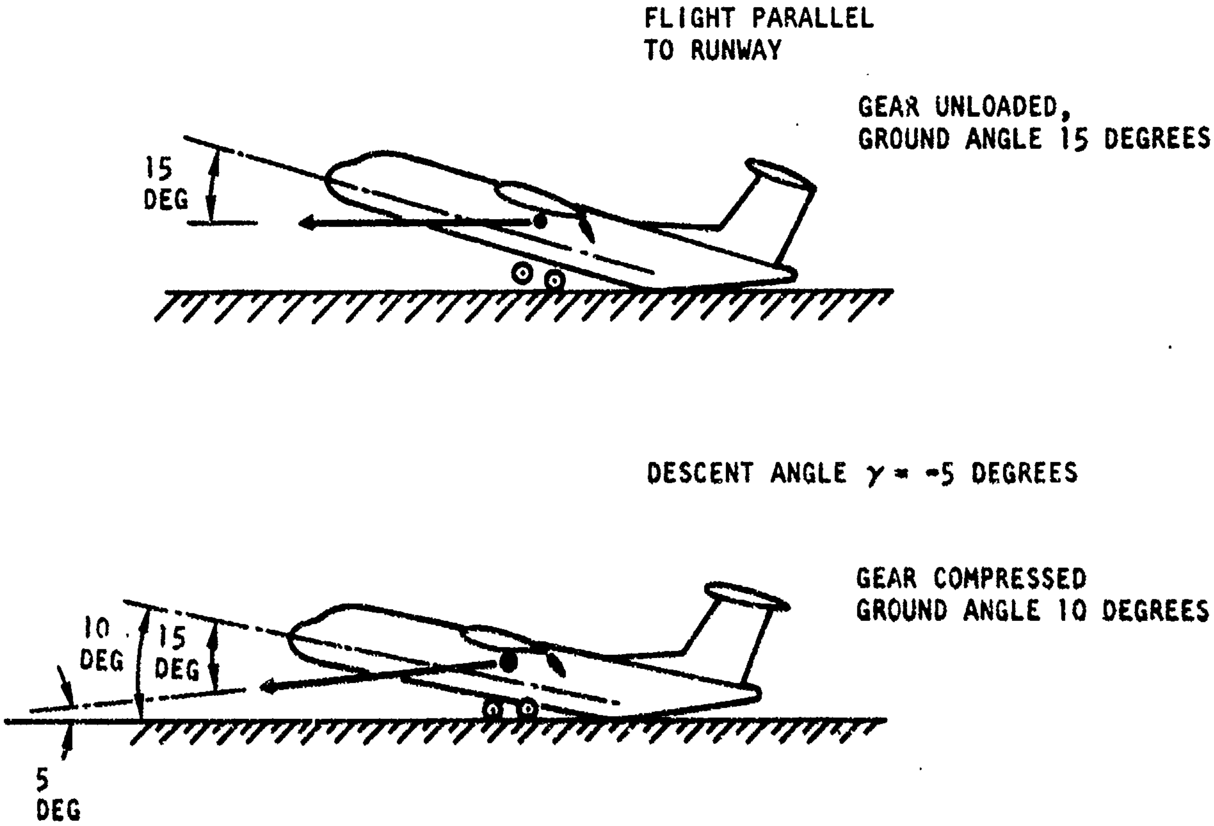


Figure 78. Ground Angle During Descent and During V_{min} Demonstration

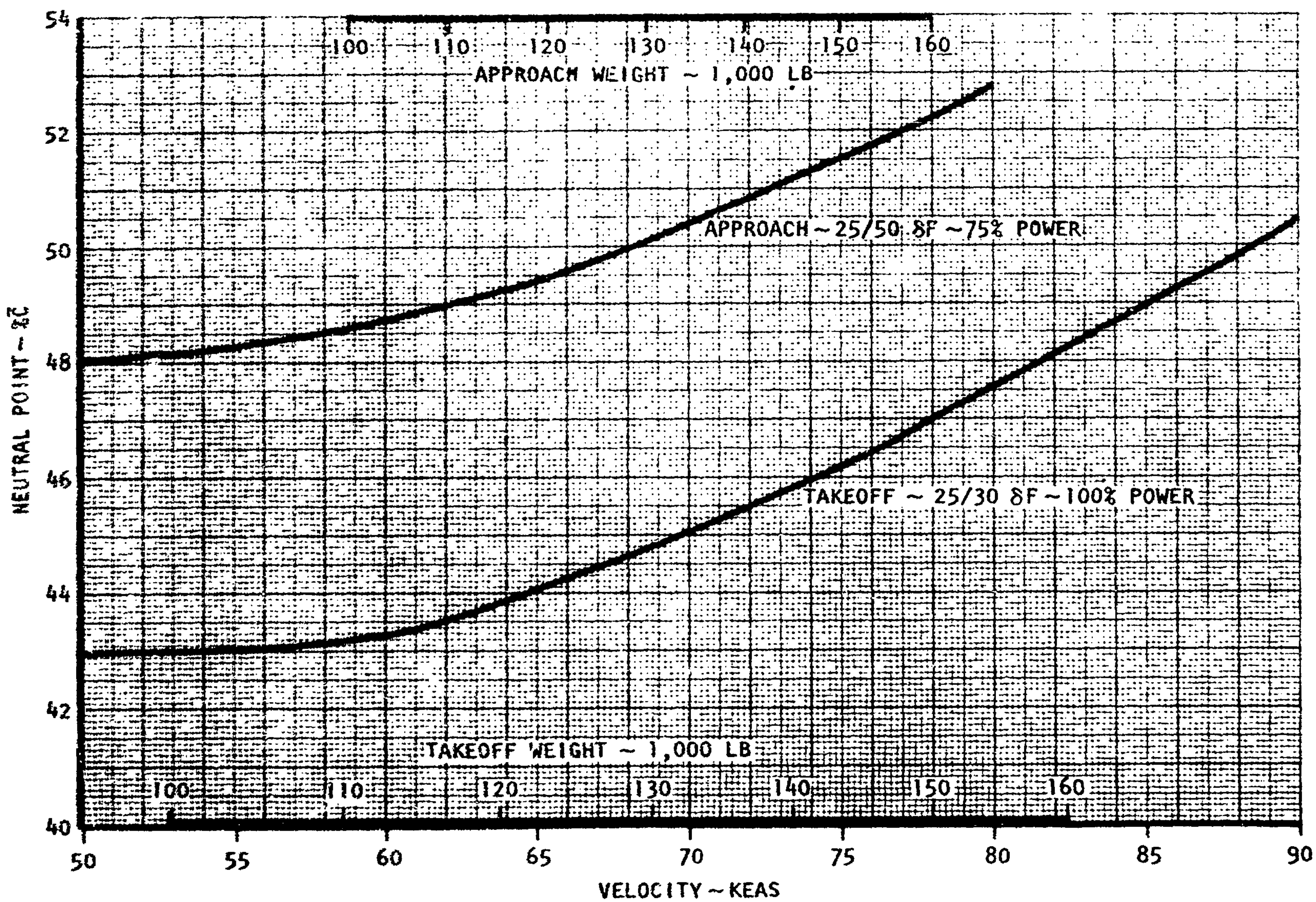


Figure 79. Aircraft Neutral Point Location

The minimum control speed in the air, V_{mc_a}, is not critical for the rudder design. Figure 80 shows the rudder deflection and the bank angle well within the limitations, even at zero sideslip angle. In this figure, the approach speed is used as a variable and aircraft weights are varied with these speeds consistent with minimum landing distances. The approach speed is used rather than the takeoff speed, because the approach speed is somewhat lower and therefore critical. The flap settings and the power level, however, are those for waveoff. The flap settings are indicated in the figure.

The flap setting has a strong influence on the yawing moment from the engine failure. An increase in flap angle increases the obstruction that is placed into the exhaust of the engines and this results in a lesser yawing moment. This decrease was determined from wind tunnel data and is shown in figure 81, nondimensionalized by the yawing moment existing at zero flap deflection

Minimum control speed characteristics on the ground, V_{mc_g}, are presented in figure 82, and are compared with the failure speed, V_F, for balanced takeoff distances. It is seen that the minimum control speed is not limiting the takeoff decision speed for gross weights above 140,000 pounds.

The minimum control speed is determined from figure 82, using the yawing moment from the rudder and the nose gear steering. The "total" yawing moment, which includes rudder, nose gear steering, and differential braking, is not used because braking precludes continuation of the takeoff. Maximum friction coefficient used for the nose gear in sideways direction is $\mu = 0.30$. The required yawing moment is derived from an outboard engine failure. It is assumed that no external blowing of the flaps takes place during the ground run in order to preserve a maximum of forward acceleration.

The crosswind landing capability is presented in figure 83 where the available sideslip capability is compared with the sideslip needed to overcome a 20 and 30 knot sidewind at a 90-degree angle to the flight path.

The sideslip capability of the aircraft is obtained directly from wind tunnel data without Reynolds number corrections as presented in figure 84. It is seen that a maximum sideslip angle of 27.5 degrees can be obtained in the power-off case; however, power effects increase the directional stability of the aircraft without increasing the rudder effectiveness. This reduces the maximum sideslip angle with power, a subject in which further research is required.

Yaw acceleration characteristics with all engines operating and with the critical engine failed are shown in figure 85. The data are shown in terms of the yaw angle reached after 1 second following a step rudder input.

NOTE: MIL POWER ON REMAINING ENGINES

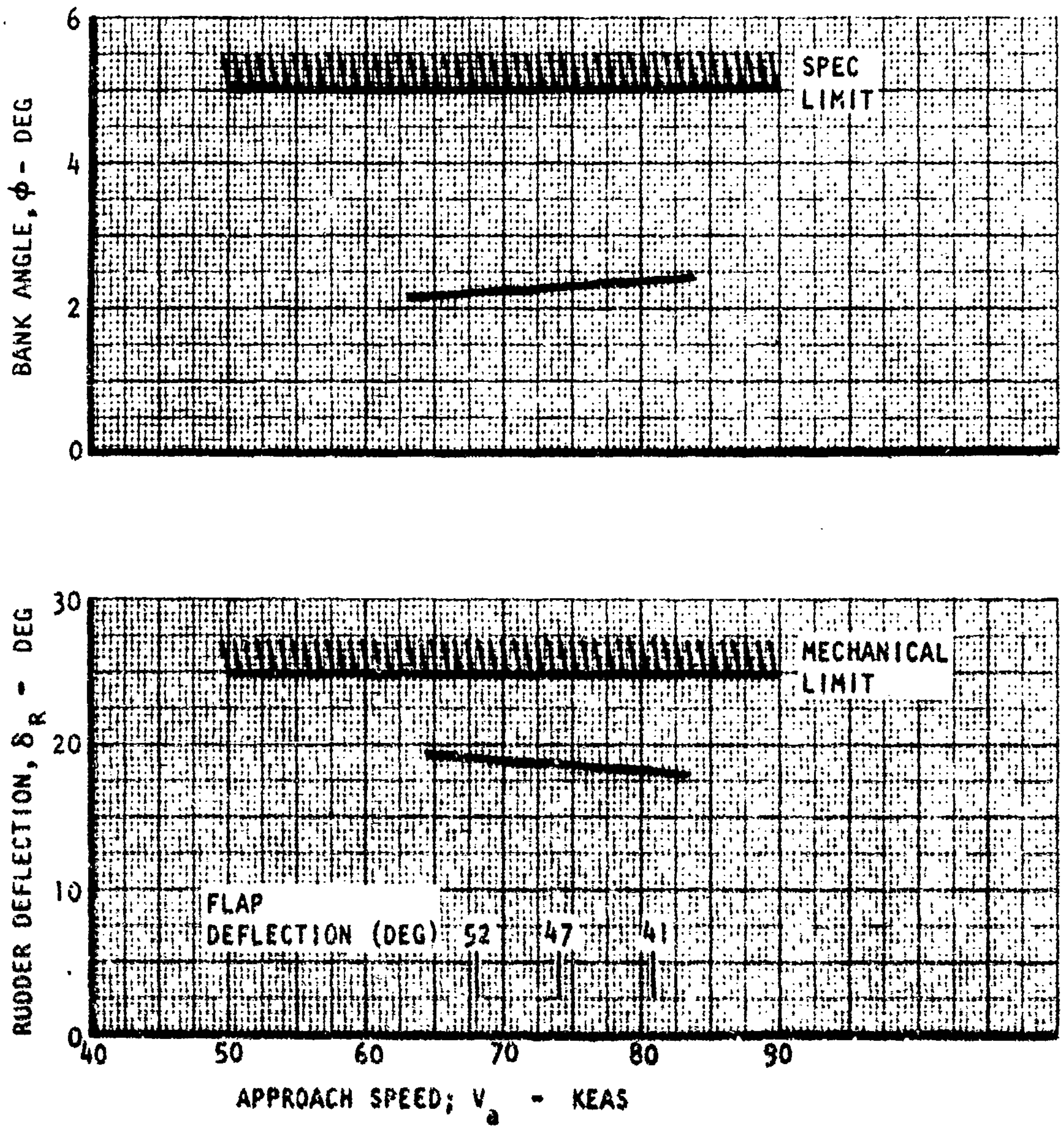


Figure 80. Rudder and Bank Angle Required at Zero Sideslip and Critical Engine Failed

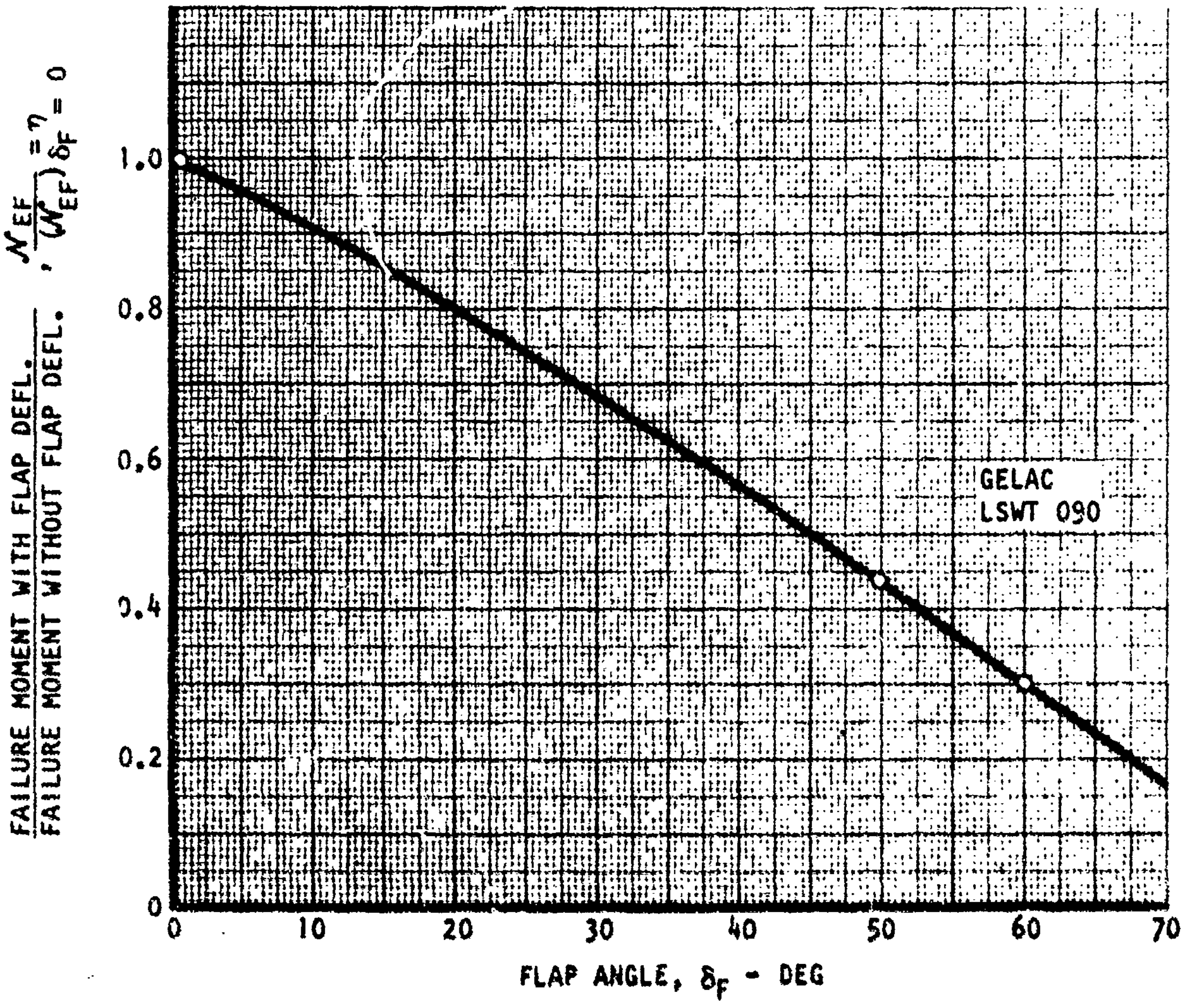


Figure 81. Effect of Flap Deflection on Yawing Moment Due to Engine Failure

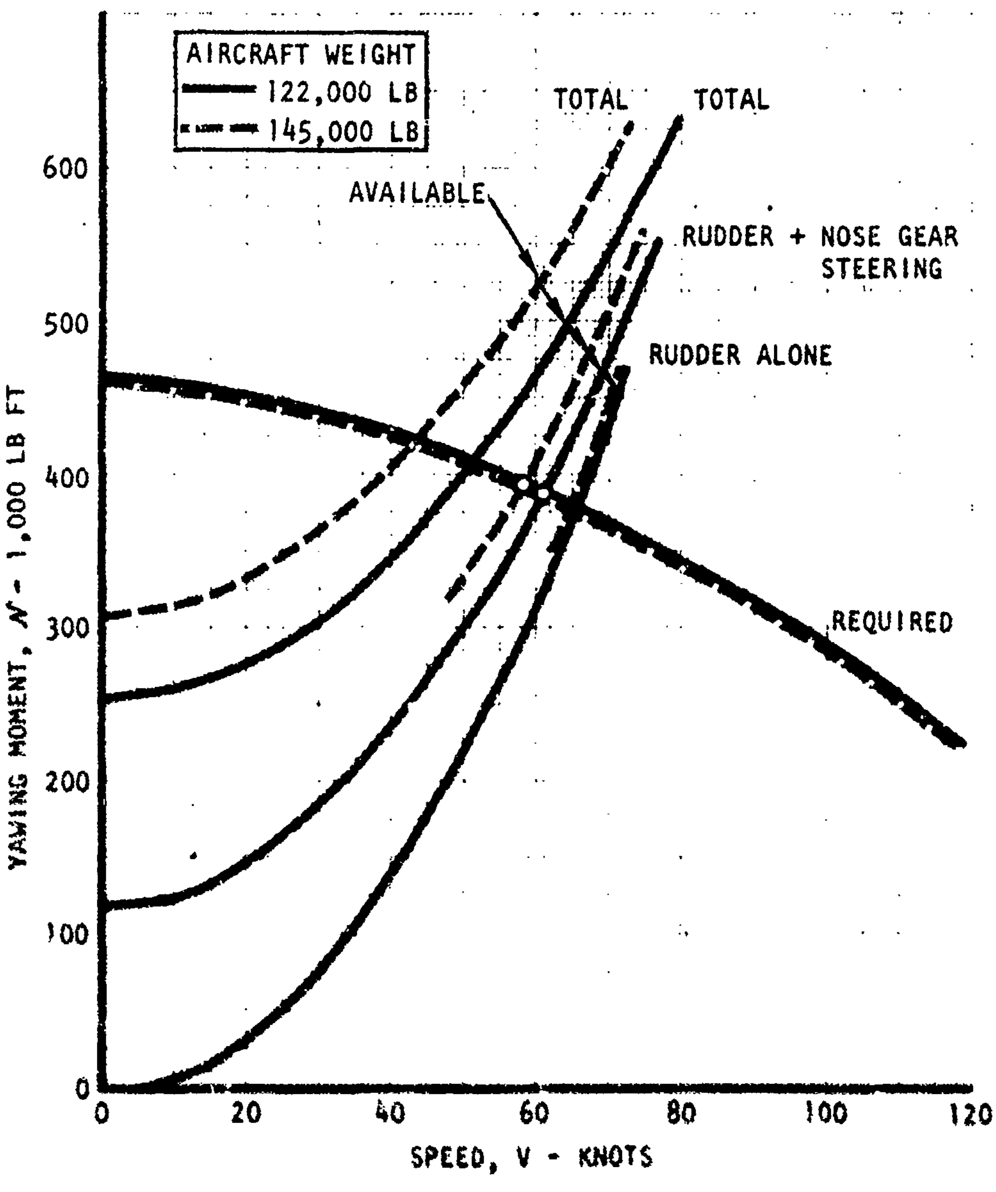


Figure 82. Determination of Minimum Ground Control Speed

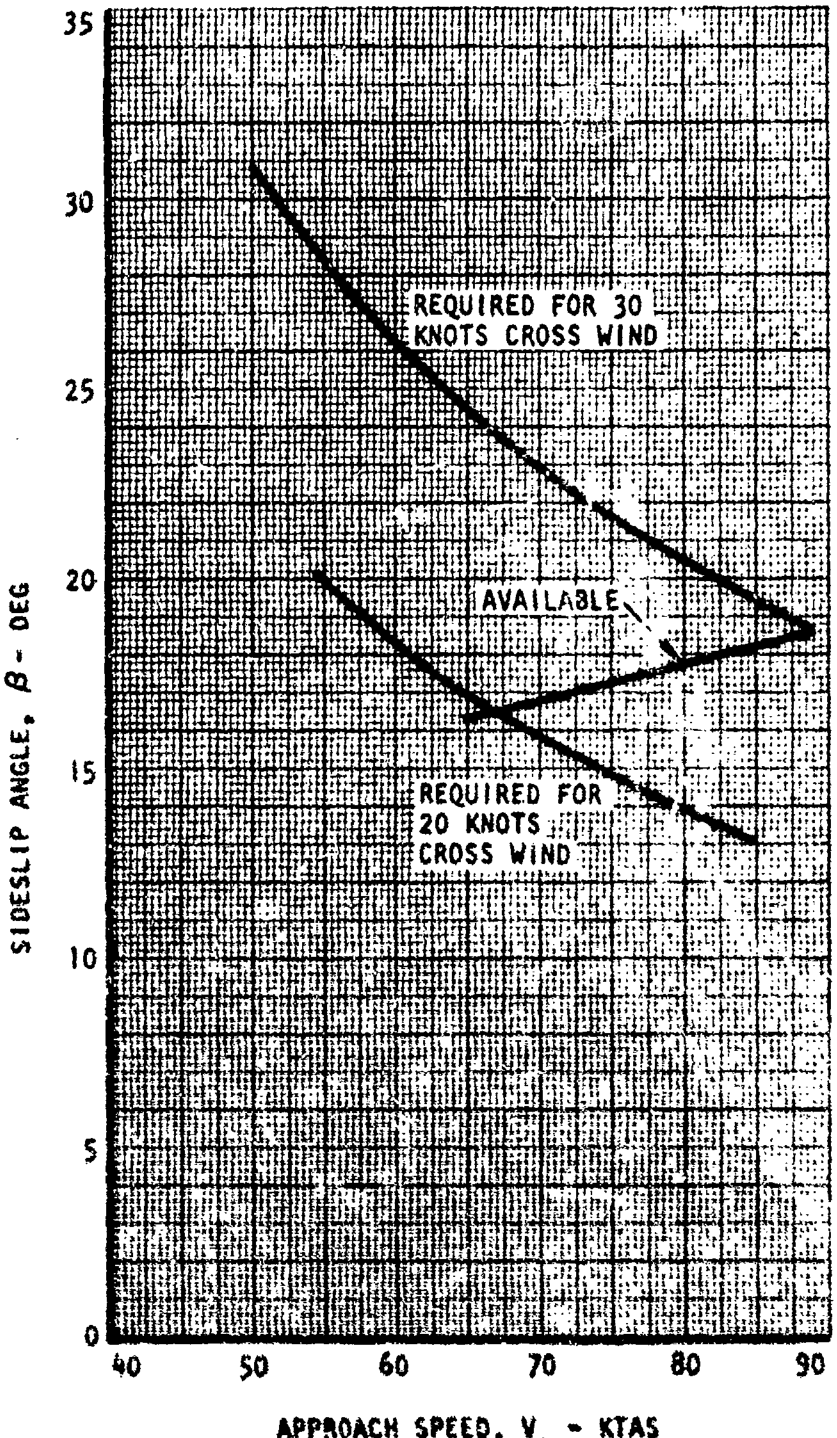


Figure 83. STOL Crosswind Landing Capability

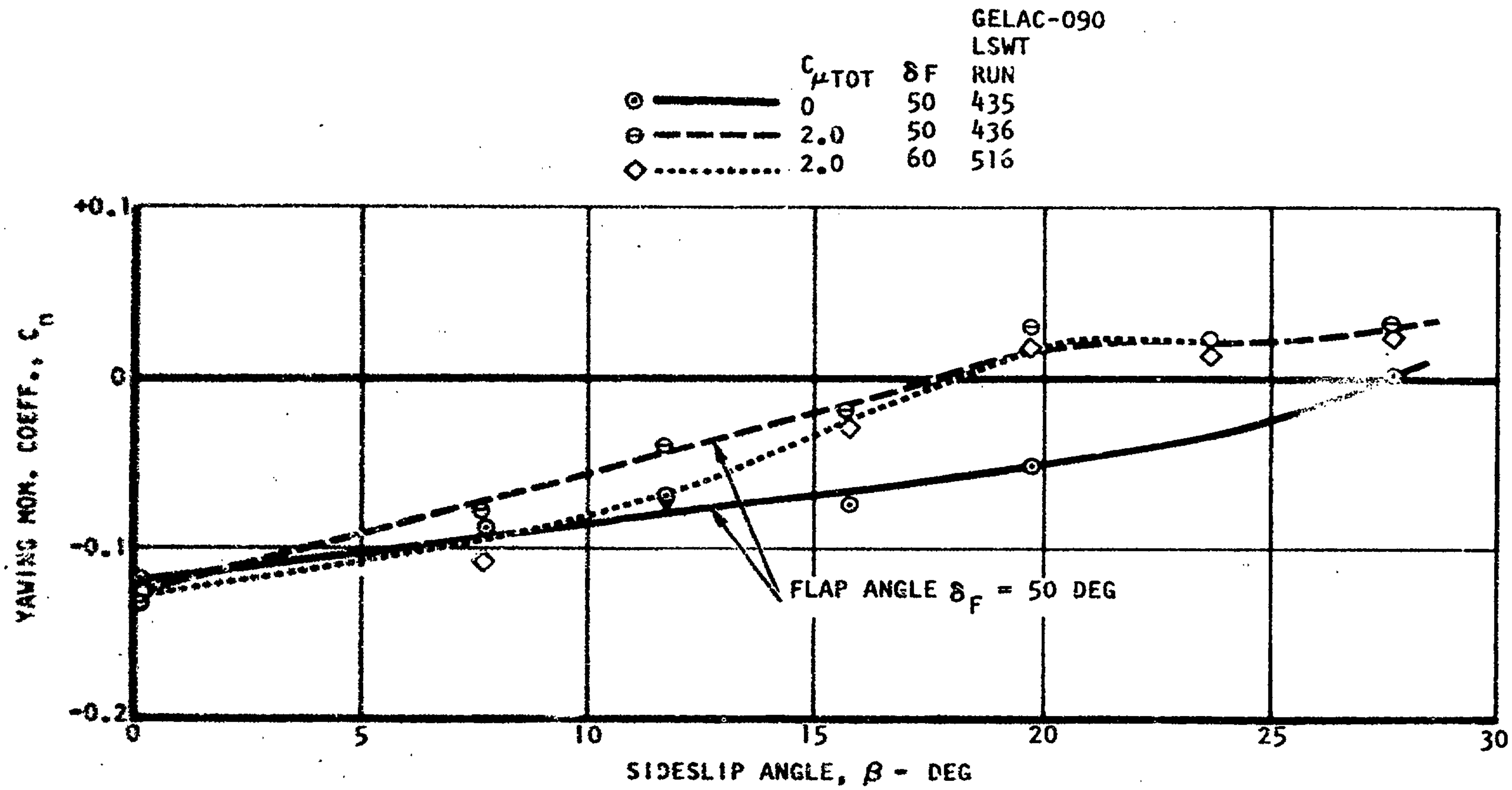


Figure 84. Yawing Moment Versus Sideslip With Full Rudder Deflection

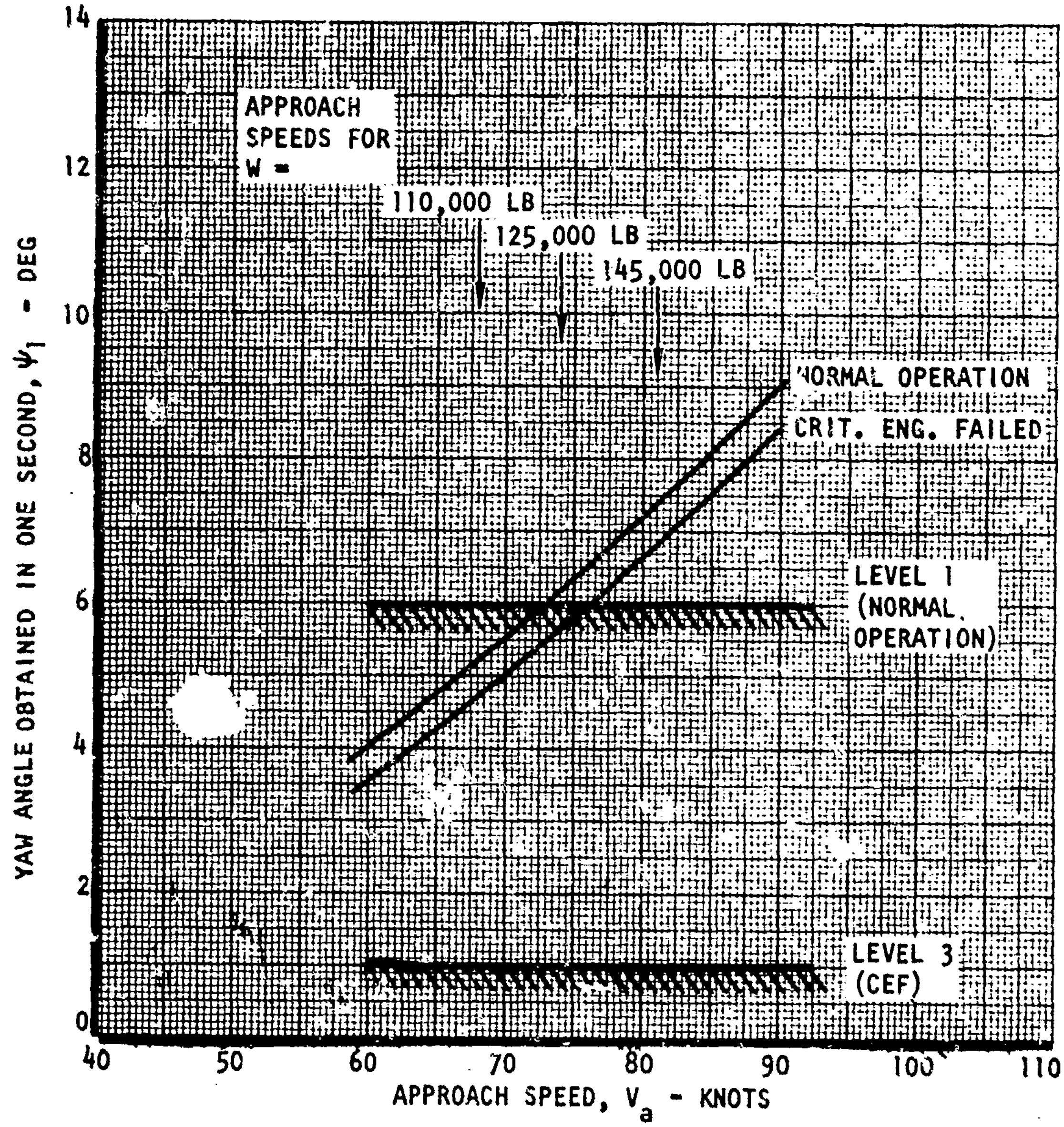


Figure 85. Yaw Acceleration Characteristics in the STOL Mode (Landing)

The requirements of MIL-F-83300 are met at speeds above 75 knots, i.e., for normal approach speeds at weights greater than 120,000 pounds.

With the critical engine failed, the yawing moment for three engines is computed using the thrust reduction with flap angle as shown in figure 81, and using a flap angle consistent with approach since the requirements are intended for decrabbing in crosswind landing (δ_F = 73 to 79 degrees for gross weight varying from 110,000 pounds to 145,000 pounds). Damping characteristics used are varying from $I_{zz}/(dN/dr)$ = 5.3 for low weights to 3.8 for high weights.

6.3 FLYING QUALITIES

For dynamic flying qualities, refer to volume V. A selection of most pertinent characteristics are included here in figures 86, 87, and 88.

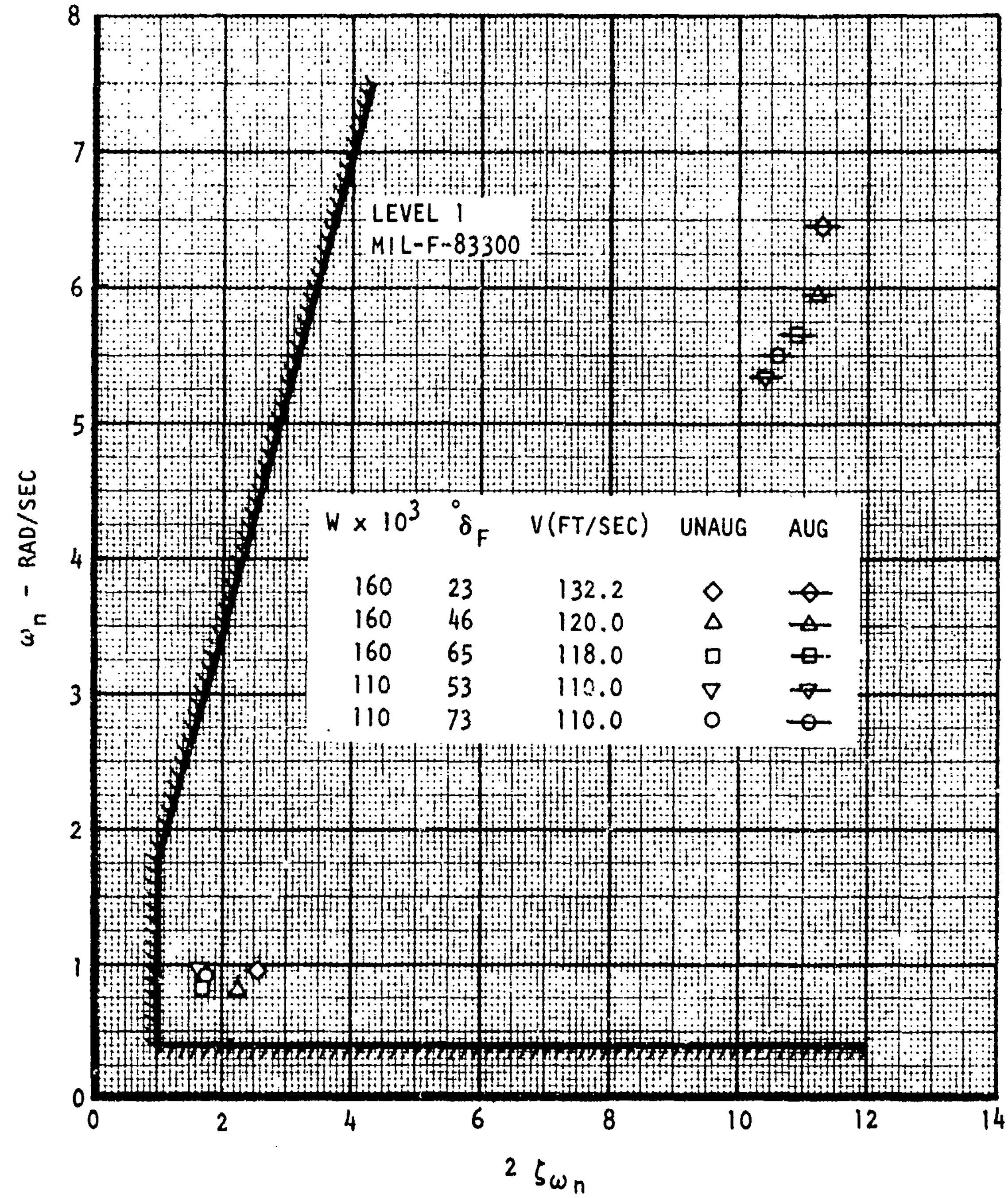


Figure 86. Longitudinal Short Term Dynamic Characteristics (Augmented Versus Unaugmented)

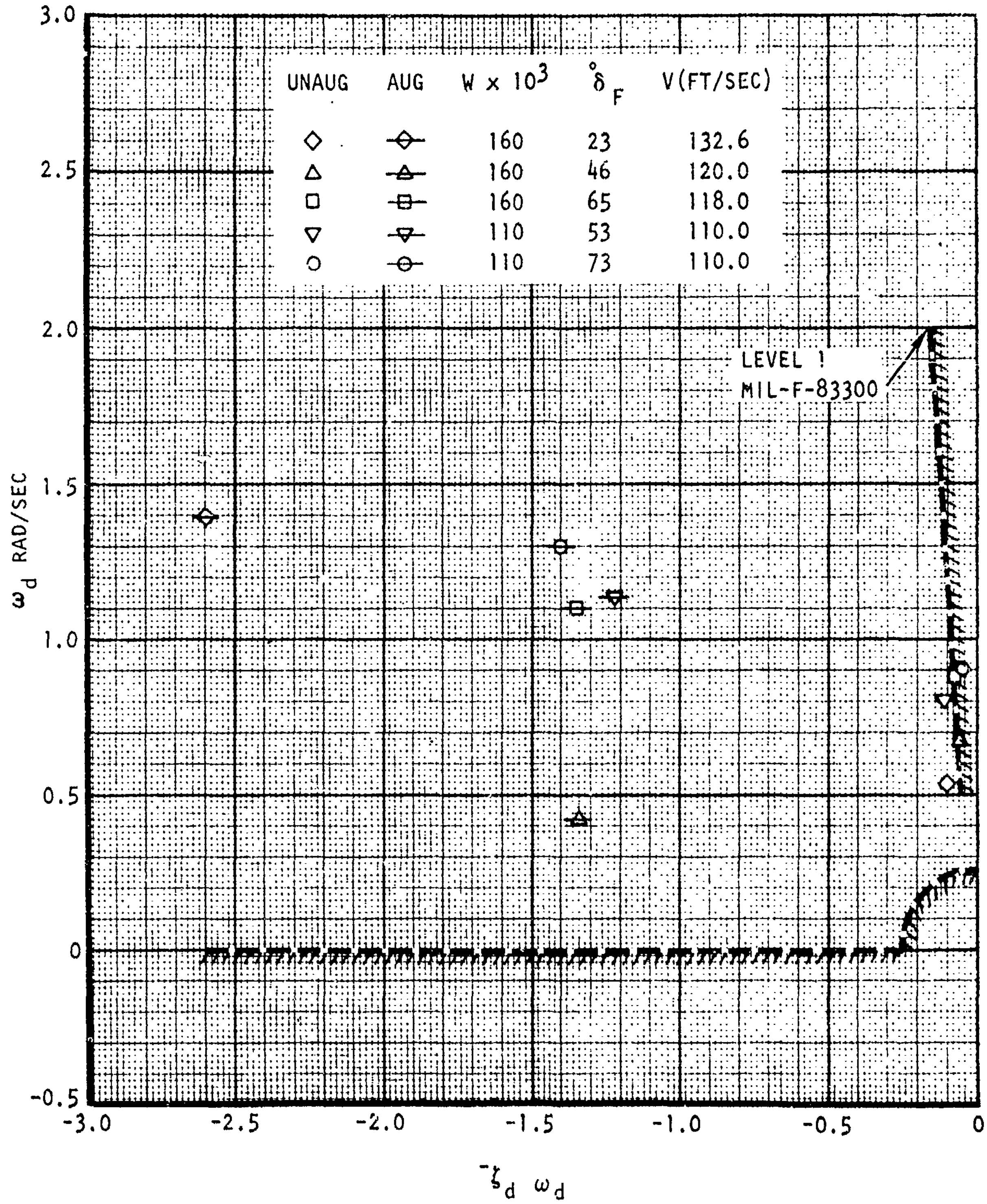


Figure 87. Dutch Roll Dynamic Characteristics (Augmented Versus Unaugmented).

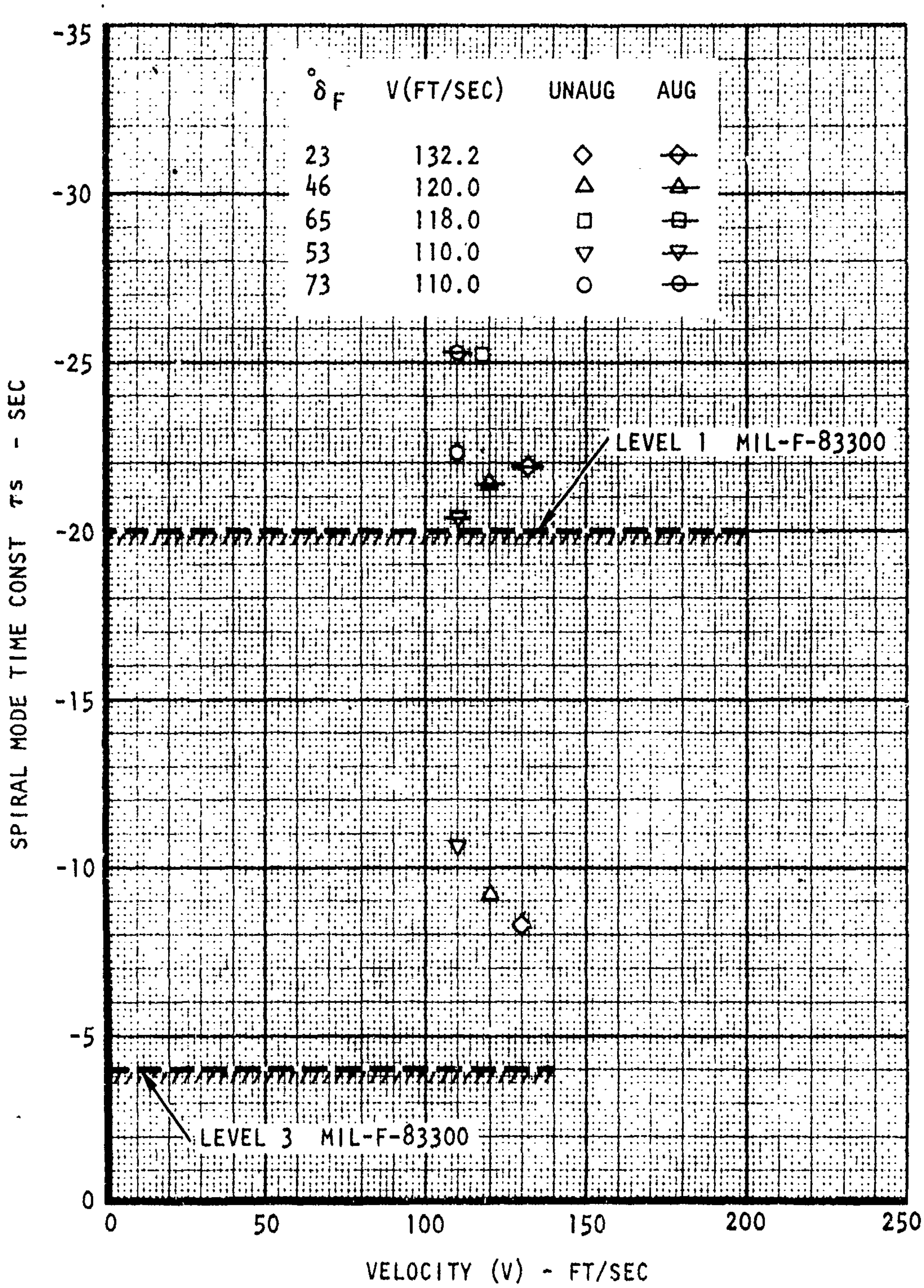


Figure 88. Spiral Mode Time Constant (Augmented Versus Unaugmented).

SECTION VII

PROPULSION SYSTEM

7.1 CANDIDATE ENGINES

A review of the STOL transport candidate engines was made for the part 2 refined configuration design. The engine companies have provided data for candidate engines, shown in table XIV, in supporting NR in this program. The existing engines would have low cost and early availability, but their weight, size, and performance do not appear attractive for future medium STOL transport designs. A derivative engine, defined as a modification of an engine under development for a weapon system using the core engine as the basic gas generator, would require funding for fan and low-pressure compressor development, but offers a much better state-of-the-art propulsion system. A new engine would incorporate more advanced technology and could be sized to match the vehicle requirements, but requires a new development program.

For the EBF configuration, a mixed-flow engine installation results in a more simple installation, regarding thrust deflection into the flap system, than a separated flow engine. Therefore, mixed-flow engines are preferred for consideration, although most turbofans with pressure ratios above 1.4 could operate as a mixed-flow engine, with only minor design changes. Another consideration for engine cycle selection is that the engine noise levels must be low enough to meet the study requirements. It is desirable to have a pressure ratio below approximately 1.6 to attain noise levels below 115 PNdb without excessive treatment. The pressure ratio range selection of 1.4 to 1.6 would be compatible to a bypass ratio range of 5 to 8. Studies have shown that varying the bypass ratio of the turbofans in this 5 to 8 range, with similar cycle characteristics, does not have an appreciable effect on the overall vehicle range or weight trade studies.

The vehicle size, flight requirements, and thrust-to-weight ratio dictates the engine thrust rating necessary to perform the mission. The preliminary evaluations indicated that the engines would need a takeoff rating of 15,000 to 35,000 pounds of thrust to meet the trade study requirements. The mixed-flow GE13/F6A engine, rated at 22,000 pounds of thrust, was selected initially as an engine cycle that satisfied the basic requirements and was representative of a new engine if the thrust-to-weight was increased. Another factor of engine choice was that sufficient data were available for this engine, during the early phase of the program, to use for vehicle evaluation. The engine uses the F101 engine core that is being developed for the B-1 aircraft. Shortly after the GE13/F6A engine selection was made, GE informed NR that an uprated version, designated the GE13/F10, had been proposed as the derivative candidate for STOL transport programs. The only basic difference between the engines

TABLE XIV. TYPICAL CANDIDATE ENGINES (SEA LEVEL, STANDARD DAY RATING)

Mfg	Model	$F_{N_{MAX}}$	T/W	BPR	FPR	OPR	Exhaust Flow	Fan Drive	Core Engine
Existing									
P&WA	JT8D-15	15,500	4.68	1.03	1.97	17.9	Sep	Direct	J52
P&WA	TF33-P7	21,000	4.52	1.25	1.9	16.0	Sep	Direct	JT3C
GE	TF34-GE-2	9,280	6.53	6.23	1.5	20.5	Sep	Direct	Scaled T-64
Allison	TF41	15,000	4.54	0.76	2.1	20.0	Mixed	Direct	RB168-25
Derivative									
P&WA	STF392A	24,054	6.38	7.92	1.4	18.0	Sep	Geared	JTF22
P&WA	STF392B	23,144	5.95	7.6	1.4	18.0	Sep	Direct	JTF22
P&WA	STF392E	31,368	6.60	8.17	1.4	24.0	Sep	Geared	JTF22
P&WA	STF402	19,751	7.60	5.0	1.6	18.0	Mixed	Direct	JTF22
P&WA	STF405	20,000	4.65	5.5/1.17	1.2/2.5	17.0	Sep	Geared	JTF22
P&WA	STF412	16,138	7.25	3.7	1.75	16.0	Sep	Direct	JTF22
P&WA	STF419	20,000	6.41	12.4/0.62	1.18	23.6	Sep	Geared	JTF22
GE	F101/F13	16,150	5.50	2.0	2.26	26.5	Mixed	Direct	F101
GE	GE13/F6A	22,000	6.52	6.2	1.46	24.6	Mixed	Direct	F101
GE	GE13/F7A	21,750	6.03	6.7	1.38/2.1	24.0	Mixed	Direct	F101
GE	GE13/F8A	14,030	4.25	2.07	3.05	24.5	Sep	Direct	F101
GE	GE13/F10B	24,000 (103°)	7.0	6.50	1.45	24.5	Mixed	Direct	F101
New									
P&WA	PW-1D	24,450 (103°)	7.09	6.0	1.5	18.5	Sep	Direct	JTF22 Tech
Allison	955-B3	20,590 (103°)	7.21	5.9	1.57	22.8	Sep	Direct	GMA100
Allison	PD351-2	21,650	8.52	5.25	1.65	24.8	Sep	Direct	GMA100
Allison	PD351-5	21,650	8.33	8.0	1.45	21.8	Sep	Direct	GMA100
Allison	PD351-7	21,650	8.77	3.0	1.8	24.7	Mixed	Direct	GMA100
Allison	PD351-72	25,165	6.37	14.22	1.26	16.7	Sep	Geared	GMA100

is that the GE13/F10 takeoff power rating is increased to 24,000 pounds of trust (flat rated to 103° F hot day) for an additional 50 pounds in weight. The GE13/F10 derivative engine uses present design technology with a supersonic engine core to achieve a basic thrust-to-weight ratio (T/W) of 7:1. The engine companies have agreed that, when new technology is incorporated to convert a derivative engine to a new subsonic engine, there would be an improvement in weight at the expense of development time and cost. GE has proposed growth versions of their derivative engine with a T/W as high as 9:1 and would estimate that a T/W of 8:1 for a new GE13/F10 type engine could easily be achieved without the use of composite materials. A weight scale factor exponent of 1.25 is used to establish a reasonable weight for a scaled size new engine which provides a T/W less than 8:1 when the size is increased from the basic GE13/F10 rating.

7.2 PERFORMANCE DATA

A scaled GE13/F10 turbofan with improved thrust-to-weight technology was selected as the new engine to be used for the part 2 study. For all power settings other than maximum power, the GE13/F6A performance was used, since GE13/F6A performance primarily differs only in the design capability of operating as a higher maximum rating. The required engine operating conditions were established to provide performance, within the engine flight envelope shown in figure 89, for all standard and hot-day mission profiles to be evaluated in this study. The installation characteristics were also established to account for performance losses due to inlet total pressure recovery, inlet cowl additive (spillage) drag, horsepower extraction, compressor bleed airflow, and fan or intercompressor bleed airflow. The external nacelle and pylon drag is accounted for in the vehicle external aerodynamics performance. The installed performance data, except for maximum power performance, were generated by GE using their GE13/F6 engine computer program that represents actual turbofan data based on the use of the F101 engine core currently under development for the USAF. The GE13/F10 maximum power performance was generated by NR/LAD from additional data points supplied by GE.

The inlet and additive drag characteristics shown in figures 90, and 91 are derived as a function of mach number and corrected airflow divided by inlet lip area. These data were obtained from the testing and analysis of subsonic open-nosed inlets conducted by NR. The inlet selected provides a pressure recovery of over 0.98 for takeoff and over 0.99 for cruise with low distortion values. The GE13 computer program in operation does not have the capability of incorporating the additive drag corrections in the installed performance. Therefore, the corrections were calculated by NR. Detailed engine performance data are proprietary with the engine manufacturer and are not included as part of this report.

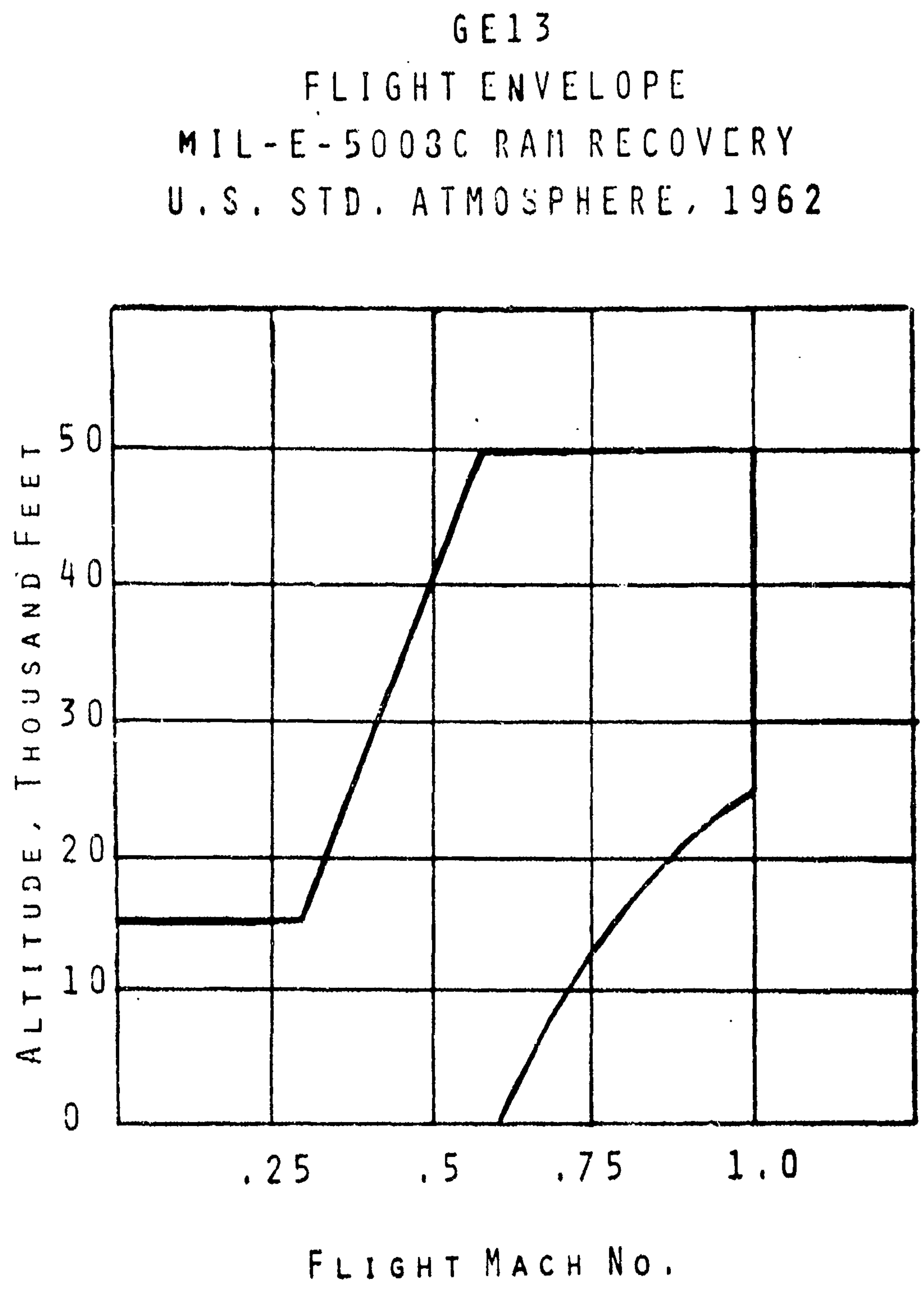


Figure 89. GE 13/F10 Engine Flight Envelope

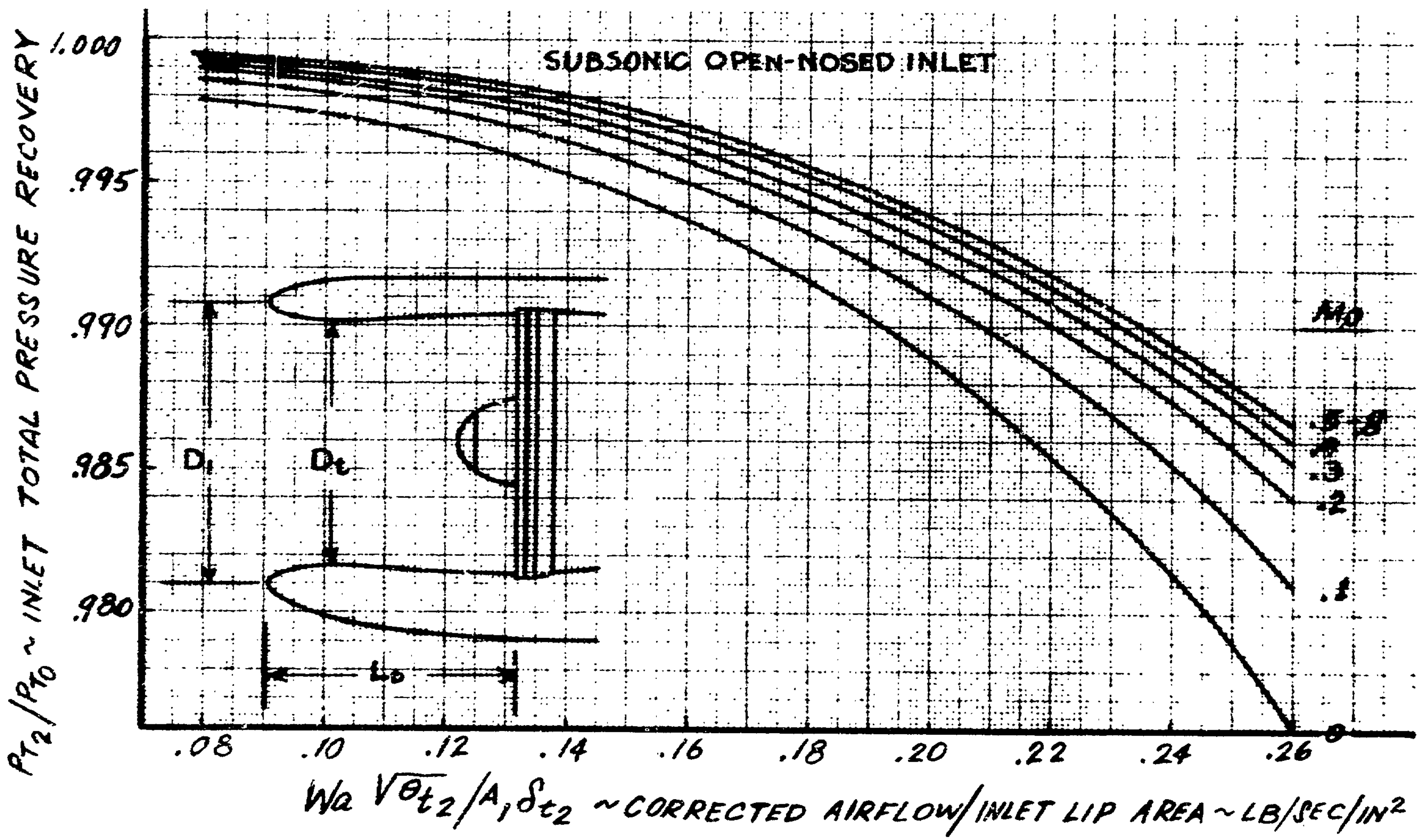


Figure 90. Inlet Recovery Characteristics

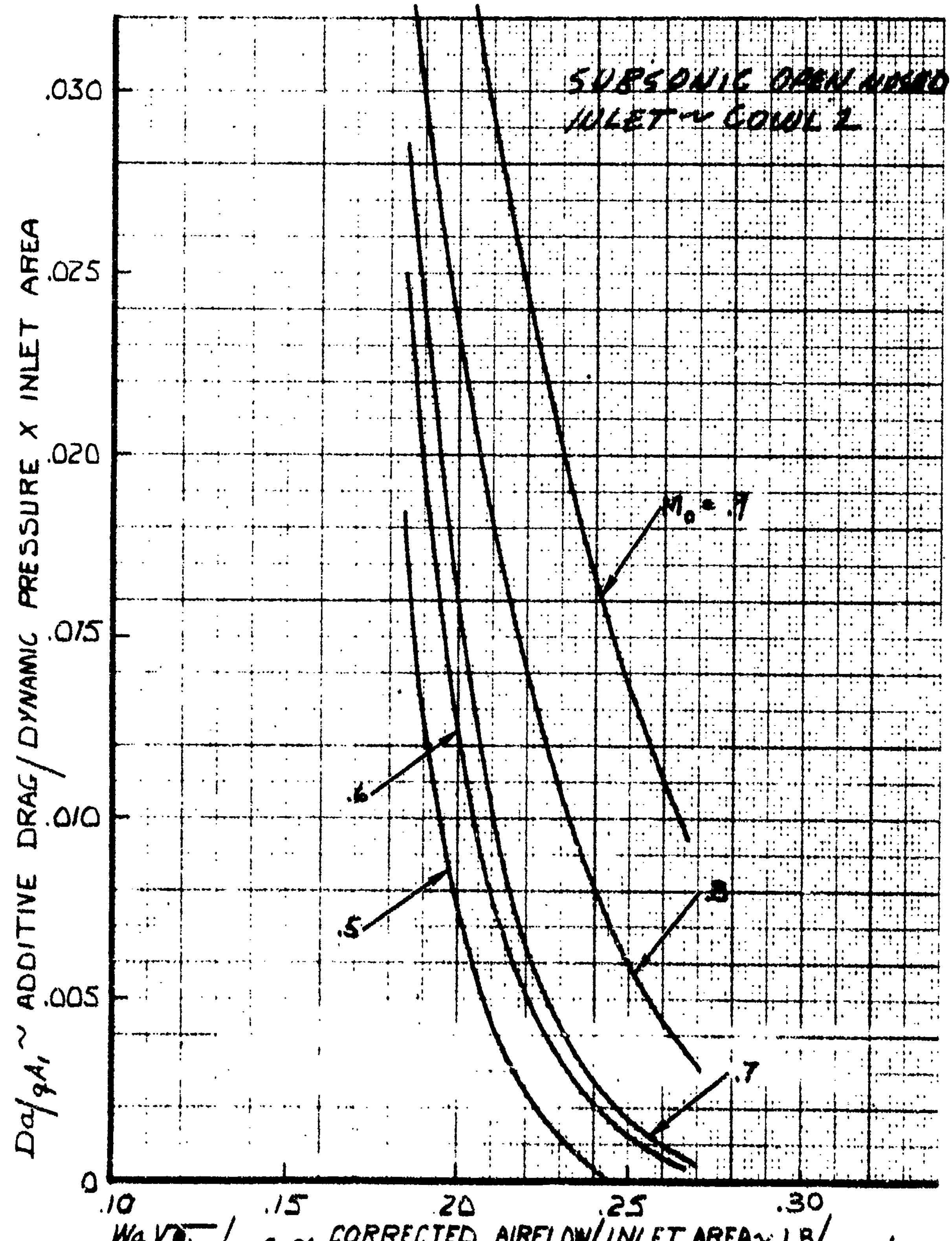


Figure 91. Additive Drag Characteristics

The horsepower extraction and compressor bleed airflow values used in installed engine performance computations were based on the vehicle design requirements. Each of the four engines has a 40 kilovolt-ampere electric generator and two 20 gpm hydraulic pumps which require a total of 80 to 90 horsepower to satisfy both the hydraulic and electric systems demands during average loads. The vehicle environmental control system requires approximately 175 to 205 pounds per minute of compressor bleed airflow throughout the different flight conditions. An average bleed airflow per engine of 1 pound per second was used to cover any additional requirements or losses.

7.3 SYSTEM DEFINITION

The inlet selected for the GE13/F10 engine installation was a simple rounded-lip, fixed-geometry, open-nosed inlet. The dimensions and performance were obtained from inlet cowl 2 data in report NA-67-550, dated 26 July 1967. This inlet was one of the configurations in the additive drag and pressure recovery data analysis from wind tunnel subsonic test conducted by NR. The model not only maintained low-additive drag and high-pressure recovery, but also indicated low distortion values of less than 1 percent for angles of attack up to 15 degrees. The inlet was matched to provide good performance during takeoff and to allow minimum spillage during cruise conditions. A bypass system was not incorporated in the design, since it would only result in a more complex installation. The inlet throat was sized for an airflow mach number of approximately 0.6 at maximum power, which provides a pressure recovery over 0.98 for takeoff. During climb, the recovery is approximately 0.985, and is over 0.99 during cruise. This sizing was found to offer a good engine airflow match for all conditions. If the inlet size were increased just to provide a slightly higher recovery for takeoff, the increase in spillage drag during cruise would tend to penalize the performance severely.

The exhaust system for the EBF vehicle requires a thrust deflector for use during takeoff and landing approach. The system will have the capability of deflecting the exhaust stream ±15 degrees from the horizontal centerline. An "eyeball" concept, shown in figure 92, was selected to be the lightest, simplest, and most rapid operating, with negligible losses during deflection or normal operation. A pneumatic actuating system operating by engine bleed airflow can change the angle of deflection to the desired position within 1 second.

The engine is installed with a cascade-type fan thrust reversing system. The cascades are located in the upper forward portion of the nacelle, just aft of the fan section. Blocker doors in the fan exhaust duct cause all the fan flow to go out the reverser cascades. The upper location of the reverser

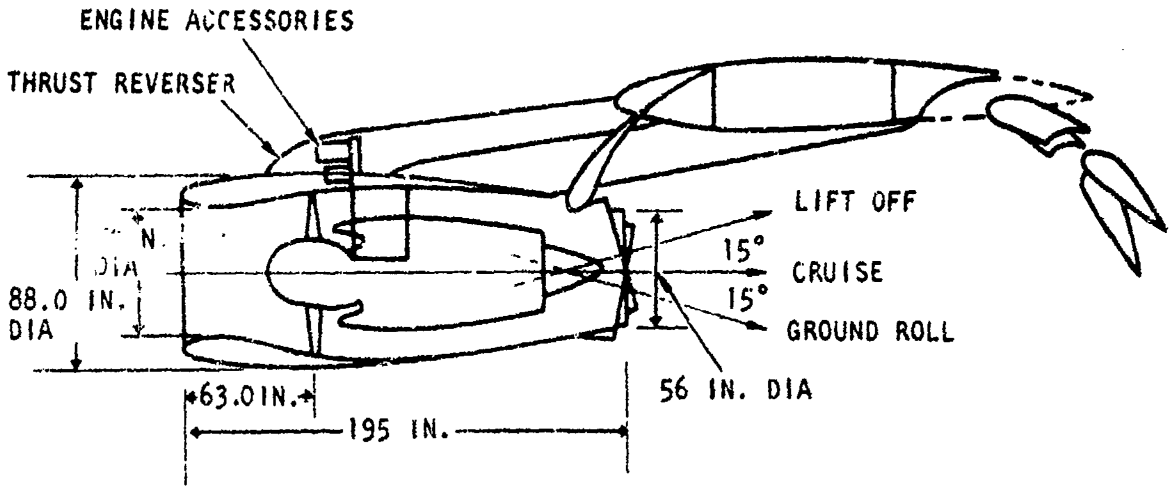


Figure 92. Nacelle Arrangement

outlet assures a minimum problem of engine ingestion while in operation. The system is also controlled by pneumatic actuators. The estimated thrust reverser effectivity, based on similar systems presently being used, is shown in figure 93. Since the core thrust is not reversed or deflected, the gas generator exhaust would produce a positive thrust. Due to the expansion of the core exhaust through the fixed common nozzle sized for the mixed flow of the high bypass turbofan, the core thrust is decayed to approximately 25 percent of the normal operating performance.

7.4 NOISE-LEVEL CONSIDERATIONS

Noise level trade studies were conducted to determine aircraft weight penalties required to achieve target noise levels of 115 PNdb sideline conditions at a distance of 500 feet. The acoustic treatment weight required to provide reasonable noise levels throughout the aircraft interior during cruise were also determined.

The external noise levels 500 feet from the aircraft consist of engine-radiated noise and flap-generated noise. The noise generated by exhaust interaction with the flap is determined from scale model data which defined noise levels with respect to the bare model nozzle noise at various flap settings and nozzle positions relative to the flap. The average noise level differences were established from the model data for flap deflection angles,

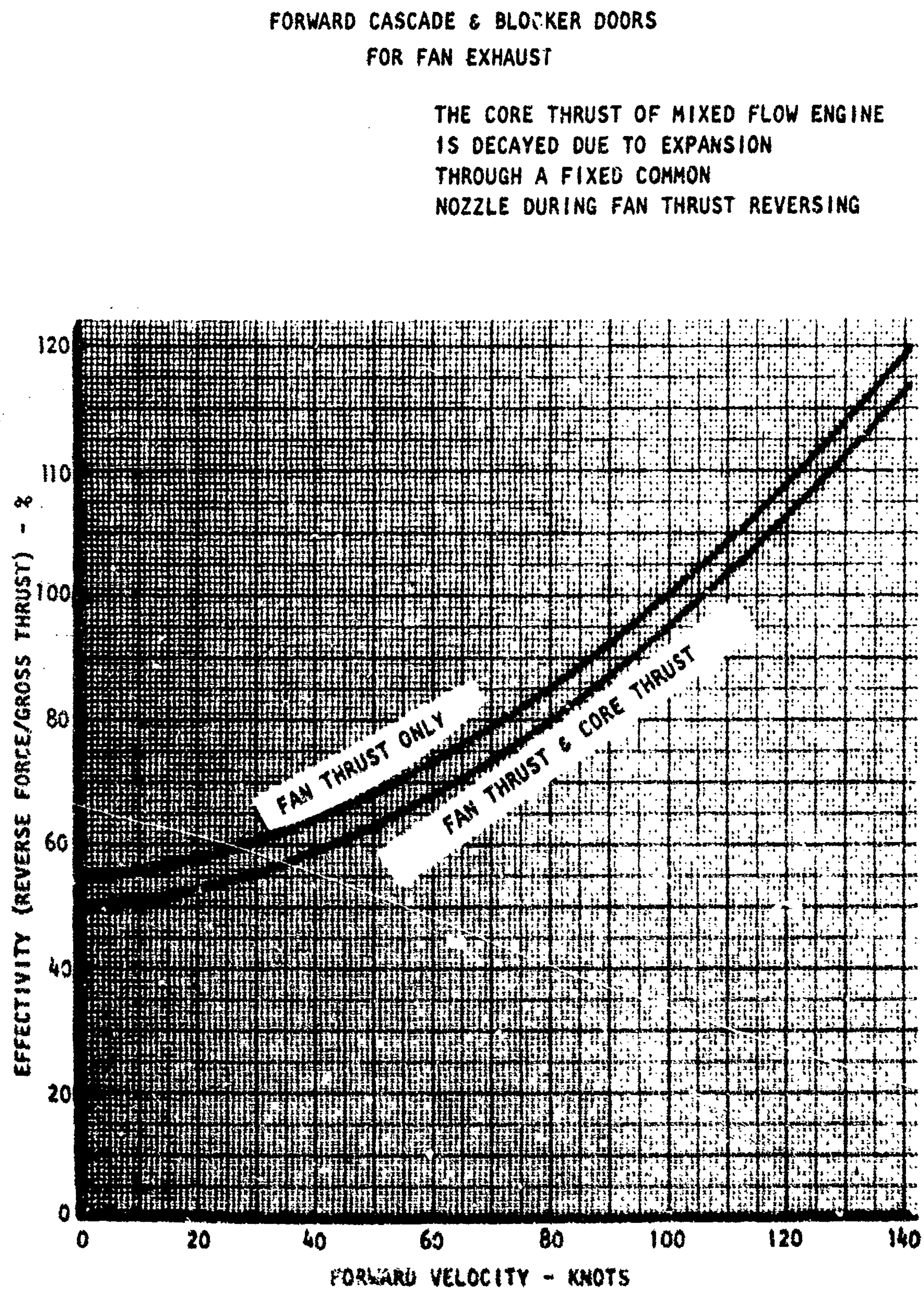


Figure 93. GE Fan Thrust Reversing Effectivity Versus Velocity

nozzle location relative to the wing and flaps, and angular orientation corresponding to the EBF aircraft configurations for sideline conditions. The bare engine exhaust noise was estimated for maximum takeoff power and extrapolated to 500 feet, employing the AIR 876 method. The scale model blown flap noise level differences relative to the bare model nozzle were added to the bare nozzle exhaust noise estimates to obtain the aircraft blown flap noise level. The blown flap noise levels are added to the engine noise to obtain the total aircraft noise levels.

The unsuppressed engine noise exceeds the blown flap noise and completely determines the total aircraft noise. The engine noise contribution to total aircraft noise decreases with increased nacelle acoustic treatment, and the blown flap noise becomes significant when engine noise is reduced to a level below 115 PNdb at the 500-foot sideline position. The blown flap noise at 500-foot sideline was determined assuming maximum engine power at liftoff and a 20-/40-degree flap angle, with the nozzle vectored for full exhaust impingement on the flap. The untreated noise level with blown flaps is 117 PNdb at 500-feet sideline. The nacelle acoustic treatment required to achieve 106 and 115 PNdb for 500-feet sideline is 4,000 and 400 pounds, respectively. These weights were extrapolated from the engine manufacturer's estimates to achieve 106 PNdb for the engine only. The total aircraft noise is achieved by attenuating the engine noise to 104 PNdb, which is added to the blown flap noise of 102 PNdb, the sum of which equals 106 PNdb. The nacelle acoustic treatment weight consists of splitters and wall treatment in the fan inlet, fan exhaust, and turbine exhaust.

The internal acoustic treatment required is 1,610 pounds. The treatment consists of fiber glass blankets applied to the entire fuselage sidewall. The treatment weights were extrapolated from previous acoustic studies based on equal treatment-weight-to-aircraft-weight ratios.

SECTION VIII

STRUCTURE DEFINITION

The structure definition for the STOL-TAI configuration incorporates designs which reflect reliable state-of-the-art structural concepts that have been proven in conventional transport aircraft systems. The structural designs and materials selected result in an airframe in which structural integrity will be maintained throughout the vehicle service life. The basic designs conform to applicable military airframe structure design criteria.

It should be noted that the analysis contained in this section are based on the part I preliminary baseline configuration. The design changes in the refined configuration, made since the structure studies were completed, do not affect the basic structural design philosophy.

8.1 DESIGN CRITERIA

Airframe structural integrity of ASD-TR-66-57 is maintained, together with the strength and rigidity requirements conforming to MIL-A-008860A series specifications. Class C transport maneuver load factors of MIL-A-008861A, table I, modified to 3 G basic design and 2.5 G maximum design gross weights, respectively, and gust criteria combine to provide a design life to 20,000 hours with 32,366 landings. Continuous turbulence criteria using the random probability analysis defined in SEG-TR-67-28 is applied with an individual airplane failure probability of 0.0005. A scatter factor of 4.0 is applied to fatigue analysis.

All known applicable airborne mission profiles (takeoffs and landings, types of missions, weight, altitude, time, speed, and other pertinent flight characteristics) and ground profiles (taxi, landing, turning, towing, and ground effects) are designed into the STAI structure.

8.2 STRUCTURE DESIGN

The structure and material combination selected is the result of static loads and fatigue computer programing which provides a realistic high degree of confidence in the STAI structural weight.

Three materials of aluminum alloy and one of titanium alloy were selected after evaluation of the mechanical and environmental properties of a range of materials including 2024-T851, 2219-T851, 7050-T7351, and 6Al-4V. These

were further studied parametrically with a final choice of 7050 aluminum alloy and a fatigue stress concentration factor $K_T = 4$. This final material and K_T factor are applied generally throughout the airplane structure to derive realistic structure weight sizing.

Aluminum alloy 7050 has been developed partially under Air Force Materials Laboratory Contract F33615-69-C-1644 and is being further developed by the aluminum industry for increasing usage while maintaining the following superior qualities:

- High mechanical properties, $\geq$ 7075-T6
- High fracture toughness
- Excellent stress-corrosion and exfoliation resistance
- Excellent fatigue performance, $\geq$ 7075 and 7059

8.2.1 FUSELAGE

This structure is designed to meet the required mission loads by design considerations for safety, maintainability, and cost-effectiveness.

The pressurized fuselage is of semimonocoque constructure and consists of a fail-safe structure that minimizes the principal safety hazard of explosive decompression. This fail-safe structure consists of:

- Nominal skin thickness optimized at 0.063 inch
- Rolled and extruded intermediate frames shear-clipped to the skin
- Thin titanium tear-stoppers between the skin and frame
- Stringers sized to tear-stopper requirements
- Crack propagation skin bays of 20-inch frame and 7-inch stringer spacing
- Dual load path type, double shear, longitudinal skin splices
- Combination of forged and built-up frame segments at main bulkheads with carry-through structure for multiple load path considerations as shown in figure 94.

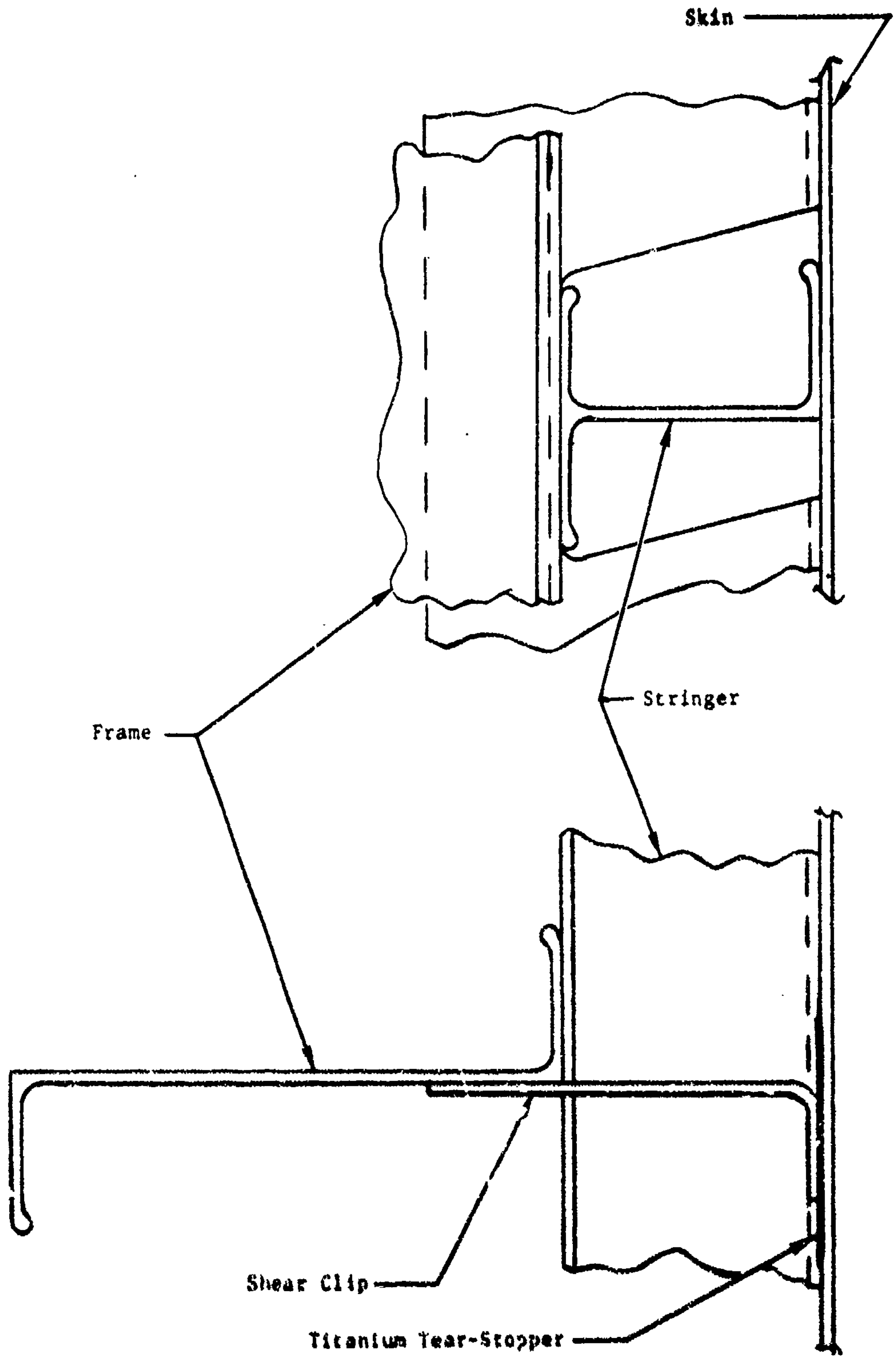


Figure 94. Typical Frame/Stringer Structure

The wing-to-fuselage juncture and the main landing gear support structure are highly loaded regions and consist of skin panels and large frames. These fatigue and stress corrosion critical areas have been recognized in the design definition. (See figures 95 and 96.)

A key element in the wing-fuselage juncture is a longitudinal titanium flexure tee sized to the cyclical fatigue life of the wing flexing at large deflections relative to the fuselage and providing an effective pin joint. Cabin pressurization loads are transferred through this tee into the wing and tend to reduce the large deflections.

The lower side frames in the main landing gear wheel well are titanium alloy forgings providing for transfer of:

- Cargo floor loads
- Cabin pressure/wheel well nonpressure
- Main landing gear
- Frame flexure

The nose landing gear wheel well is nonpressurized, as is the main landing gear wheel well, with closure frames of aluminum reacting vertically into the crew floor substructure. Both the main and nose landing gear doors and fairings are of honeycomb-stiffened construction.

The aft fuselage is a fail-safe structure designed to a mix of empennage loads, pressurization and nonpressurization, open shell at airdrop conditions, etc. This structure is a beaver-tail shape open shell supporting a ramp, aft door, pressure door, pressure bulkhead, and empennage.

The shell upper torque box, shown in figure 97, provides for torsional stability of the empennage and is reacted at the ramp hinge bulkhead with additional runout damping 50 inches forward. Large longitudinal lower side keel beams provide adequate stiffness to dampen out the dynamic excitation of an open shell. This critical condition exists when the ramp and doors are open during airdrops. The airdrop condition requires a dynamic response factor of 1.50.

The ramp provides for loading and unloading of wheeled and truck vehicles, pallets, ingress and egress of troops and litter personnel, plus aerial delivery and in-flight cargo jettison. The ramp is operated by a hydraulic actuator on each side approximately two-thirds distance aft of the hinge line. When the ramp is lowered to the ground for loading or unloading, the stored on-board, lightweight ramp toes (dock boards) are manually positioned to bridge from the aft end of the ramp to the ground.

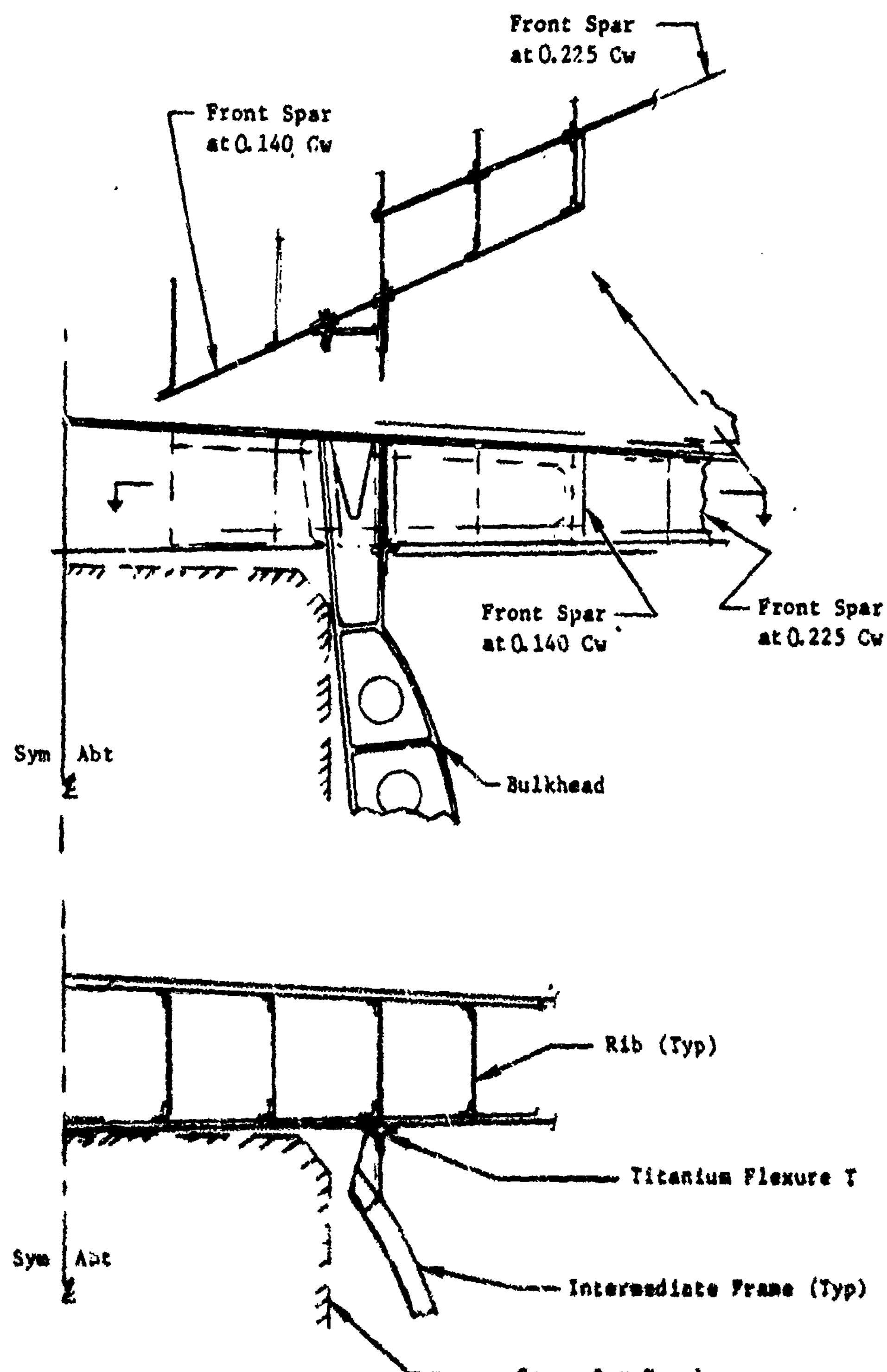


Figure 95. Wing-to-Fuselage Juncture

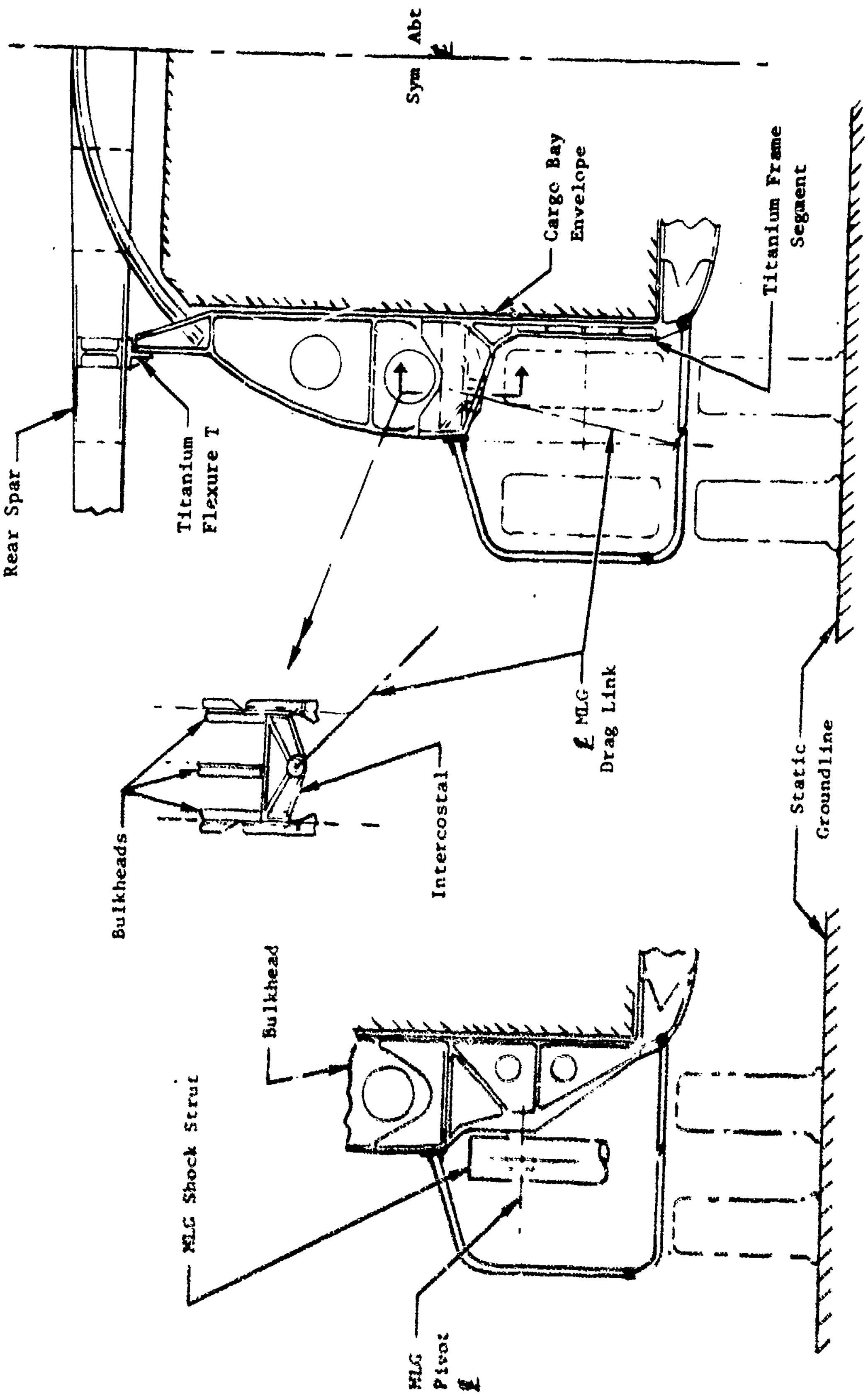


Figure 96. Landing Gear Carry-Through Structure

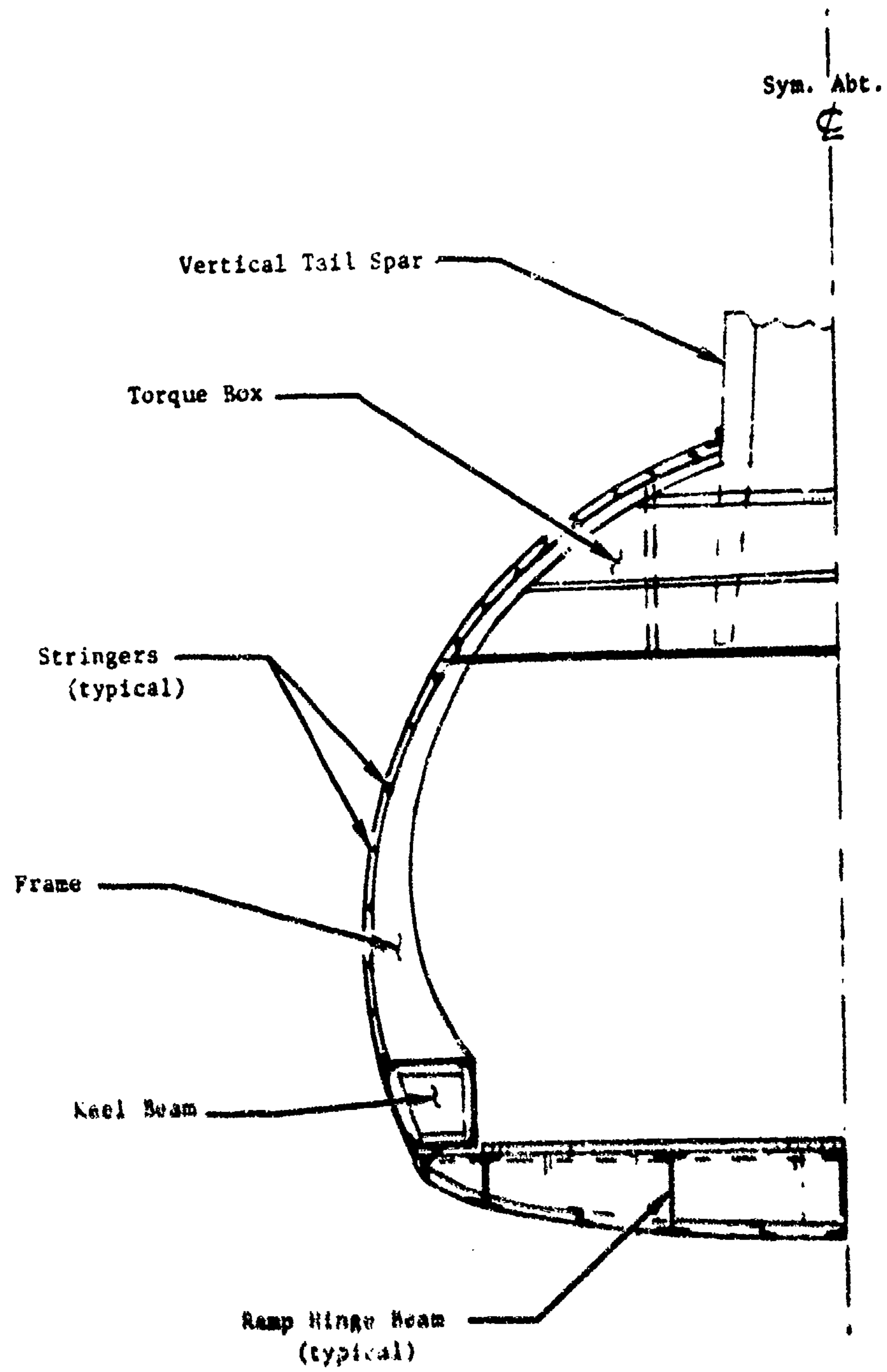


Figure 97. Aft Fuselage Section

Fore-and-aft running beams and lateral intercostals transmit fuselage torque shears, and pressure through the forward end hinges and side latches during flight when the ramp is closed. When the ramp is open, loads are transmitted through the hinges and two actuators into the fuselage frames. Zee stiffeners run longitudinally and laterally (eggcrating) to stabilize the ramp outer skin. Extruded floor planks on the ramp are easily removable to allow inspection within the ramp and replacement as required. Recessed tiedown rings and roller conveyors are integral with the ramp floor planks. These roller conveyors are flush-recessed and are flipped 180 degrees for cargo use. (See figure 98.)

The aft cabin pressure barrier consists of a fixed bulkhead with a pressure-resistant, plug-type door located at the aft end of the ramp. The pressure door is hinged at the upper edge and swings forward and upward and is flush-locked at the upper torque box when a clear rear end opening is required.

The pressure door is flat and is shear-web-stiffened vertically and horizontally (eggcrating). The stiffeners are on the aft side of the web to allow the pressure web to bear directly against the stiffeners.

The aft door retracts up and inward to provide clearance when ground loading or unloading and aerial delivery is required. This door is aft of the cabin pressure closeout door and is not considered part of the aft fuselage primary structural shell. Aerodynamic pressure and acoustic loading are design conditions for the aft door.

The construction of the aft door is aluminum alloy honeycomb sandwich; this provides a light rigid door for efficient handling of the acoustic fatigue and aerodynamic loads.

The cargo compartment floor is designed with the capability of supporting wheeled and tracked vehicles, pallets, and intermodel cargo containers to accommodate the 463L material handling and aerial delivery systems, and includes additional strength criteria capability of MIL-A-008865A.

The cargo floor comprises simple removable and replaceable extruded planks with recessed tiedown ring beams, reversible roller conveyor trays, and guide and restraining rails. Floor and frame fittings for aeromedical requirements are provided. The roller conveyors and rails are segmented for variable installations to allow any combination of mixed pallets and vehicle loads.

The longitudinal planking assembly spans across transverse floor support bulkheads at 20-inch intervals. Two longitudinal keel beams are in the floor support substructure together with the bulkheads. These keel beams resist fuselage vertical bending loads and distribute heavy concentrated floor loads to adjacent bulkheads. The floor bulkheads and keel beams are of stiffened web and cap flange built-up construction.

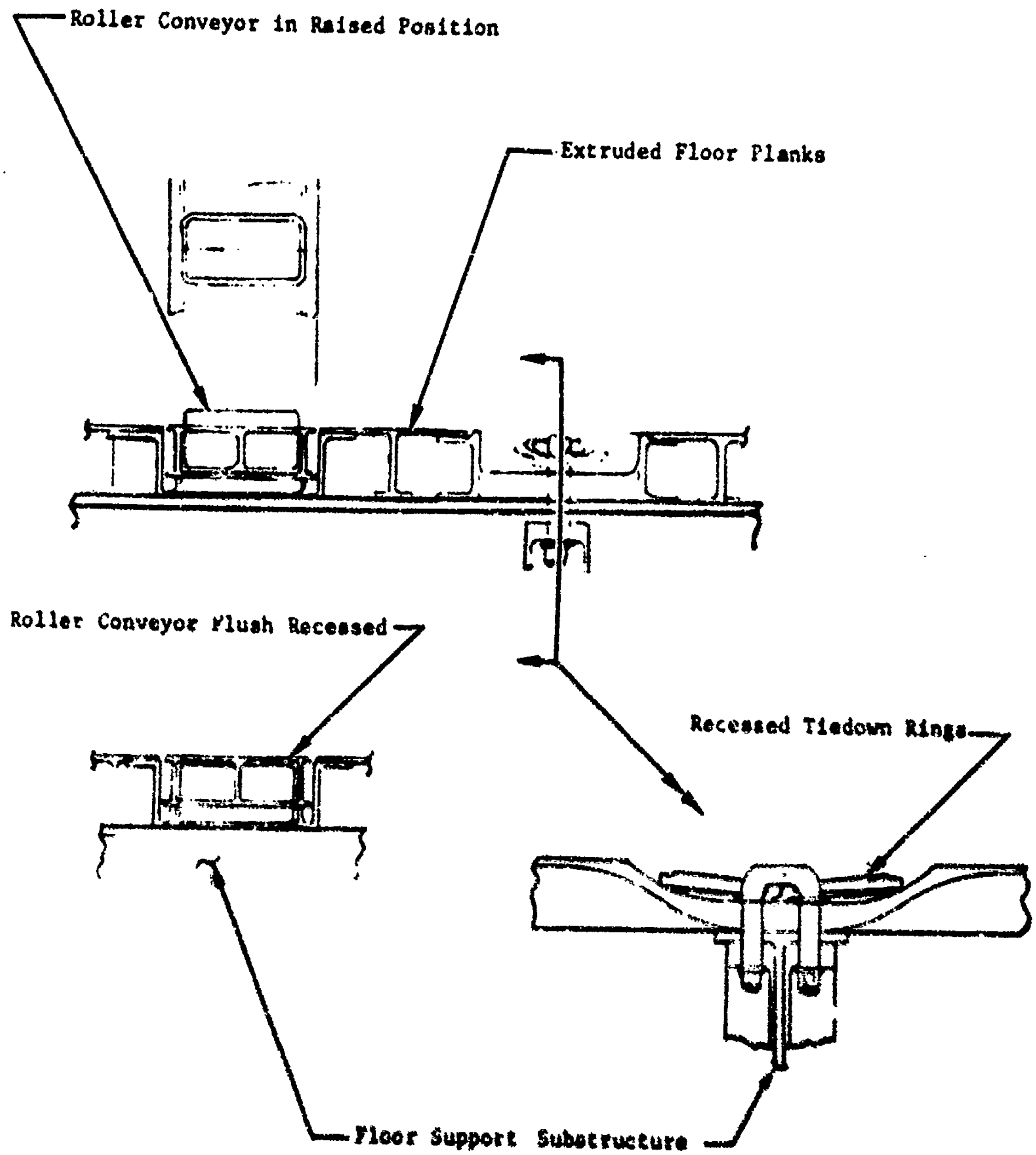


Figure 98. Cargo Floor Features

Criteria for strength design of the cargo floor include the following:

- Paragraph 3.6.3 of MIL-A-008865A

- A single-wheel axle load of 5,000 pounds using a tire size of 8-1/2 x 20 inches

- A 6,500 pounds per wheel forklift truck with solid rubber tires of 7-1.4 x 22 inches or 1,600 pounds per wheel with steel tires of 8 x 3 inches

- Floor panels, treadways, and other structure components are designed for rolling loads, during loading or unloading of cargo times a limit handling factor of 2.0 down

- Loads associated with the 463L cargo system of 10,000 pounds per pallet resulting from crash, flight, and aerial delivery systems

- Critical floor design loads include both crash and flight loads

- 300 pounds-per-square-foot limit minimum throughout entire floor with 400 pounds-per-square-foot limit on the floor beneath the wing

8.2.2 WING

The wing is fully cantilevered with two spars and multiple stringers and is a fatigue-resistant, fail-safe structure designed with corrosion prevention and ease of maintainability features.

The wing primary structure box reacts loads from leading edges, slats, double-slotted flaps, flaperons, spoilers, and engine nacelles. This structure box acts as a box beam and consists of a front and rear spar, multiple stringers, machined skins, ribs, and bulkheads. This full-span structural box also serves as an integral fuel tank and is designed to meet the requirements of MIL-F-38363A and its associated documents.

The wing structure is designed to carry a fail-safe load with the failure of any one member. These fail-safe structure features consist of:

- Multiple machined skins optimized to require variable thicknesses

- Dual load path, double-shear, spanwise skin splices

- Stringers sized for crack growth retardation

- Superior fracture toughness material

- Separate members for stringers and spar caps

- Heavy splice stringer

Wing ribs are composed of stiffened webs with deep lightening inspection holes and cap flanges. Attachment of the ribs to stringers are through forged fittings that provide column support for skin panels, react tank pressures without eccentricity, and minimize preloadings. (See figure 99.)

The skin stringer combination is sized to the primary wing bending conditions at the fatigue-resistant and fail-safe level. Upper skin thickness range is optimized at 0.163 to 0.080 inch and lower skins at 0.178 to 0.080 inch. Zee stringers 2 inches deep at 3-1/2 inch spacing vary in quantity from 34 to eight, with a thickness range optimized at 0.0[illegible] to 0.060 inch.

Full-span leading edge slats are driven by a screwjack actuation system. Two screwjacks on each slat segment are interconnected by torque tubes powered by hydraulic motors. Gearboxes located on the aft end of each screwjack transfer the rotary motion of the torque tubes to linear motion of the screwjacks for extension and retraction of the slats. Except for high angles of attack, the differential pressure (5.0 psi) acting on the slat segments in a forward direction aids slat opening. Therefore, tension loads are critical for the screwjack system during retraction of the slat. Slats are attached to the wing by curved titanium tracks supported from antifriction type bearings in the wing leading edge.

The four inboard slat segments, when extended, form an angle at the WRP of -30 degrees from the retracted position. The slat trailing edge remains in contact and forms a seal with the wing-under-slat skin during extension. In the event of an engine-out condition, mechanical linkages located between the slat front and rear beams, driven by differential motion of the screwjack actuators, permit opening the slat trailing edge to form a gap equal to two percent of the local chord. This resulting leading edge slot configuration effectively reduces the risk of a locally stalled wing due to loss of external blowing from the failed engine.

The two outboard slat segments operate in a manner similar to the inboard slats, except that they extend to an angle of -38 degrees. The inboard slat chords are 2[illegible] percent and the outboard slats are 25 percent of the wing local chord and travel normal to the wing 26.5-percent element line. The configuration of the slats is conventional formed aluminum skins, formers, spars, and trailing edge extrusion as shown in figure 100.

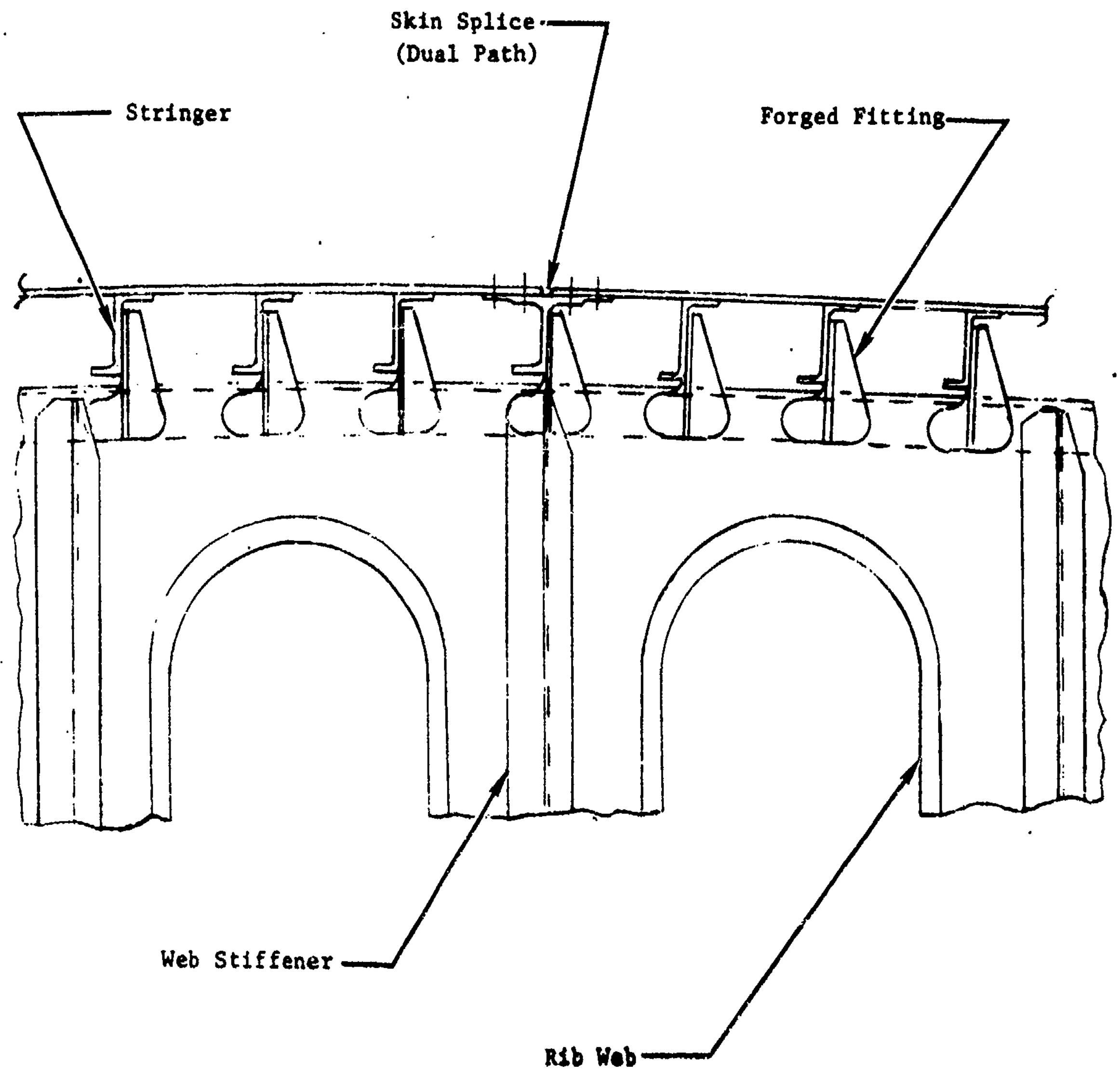


Figure 99. Typical Wing Structure

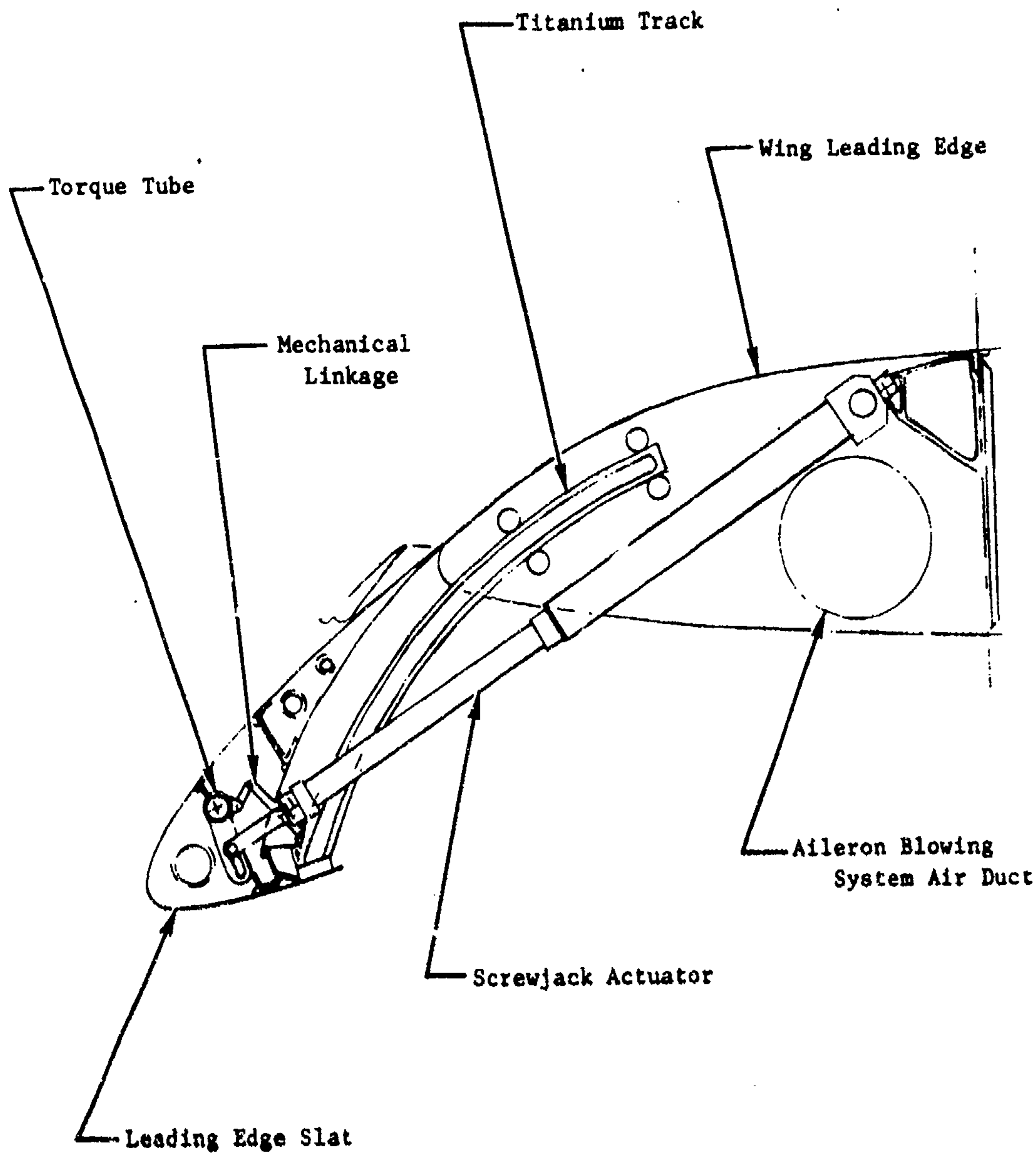


Figure 100. Wing Leading Edge Slat

The externally blown double-slotted flap is designed to operate on a simple external-type hinge incorporating a fail-safe double hydraulic actuator system. This aerodynamically faired simple external hinge system is chosen over a track/roller system for the following advantages:

- Fewer parts
- Lower maintenance costs
- Higher reliability
- Minimum weight
- Minimum construction costs
- Minimum field ground time
- Mechanical simplicity

The large double-slotted flap system is 36 percent of the wing chord and is divided into two chordwise segments. The flap system per side is divided into three spanwise segments. The forward flaps extend to a maximum of 30 degrees. The aft flaps extend to 60 degrees with the inboard one having an added articulated capability of ±25 degrees from the fully extended position for direct drag control.

The material and construction for this system is conventional state-of-the-art. As the maximum short-term surface impingement temperature is only 240° F with moderate pressures and velocities, 2024 aluminum alloy skins with 7050 internal structure are predominantly used throughout with steel fastener-bearing combinations. The forward flap is of double spar and rib construction. The aft flap is a single spar and rib construction. The ribs are sized to take the high 134-decibel sonic pressure level from the engines. The hinge fitting, support, and bulkhead are conventional aluminum alloy forgings. Flutter is not critical for design of this stiff hinge and bellcrank design shown in figure 101.

The two-part aileron is designed to operate on an efficient four-bar linkage system.

This aerodynamically faired four-bar linkage system is chosen over a track-roller system for the following advantages:

- Simplicity of parts
- Higher reliability

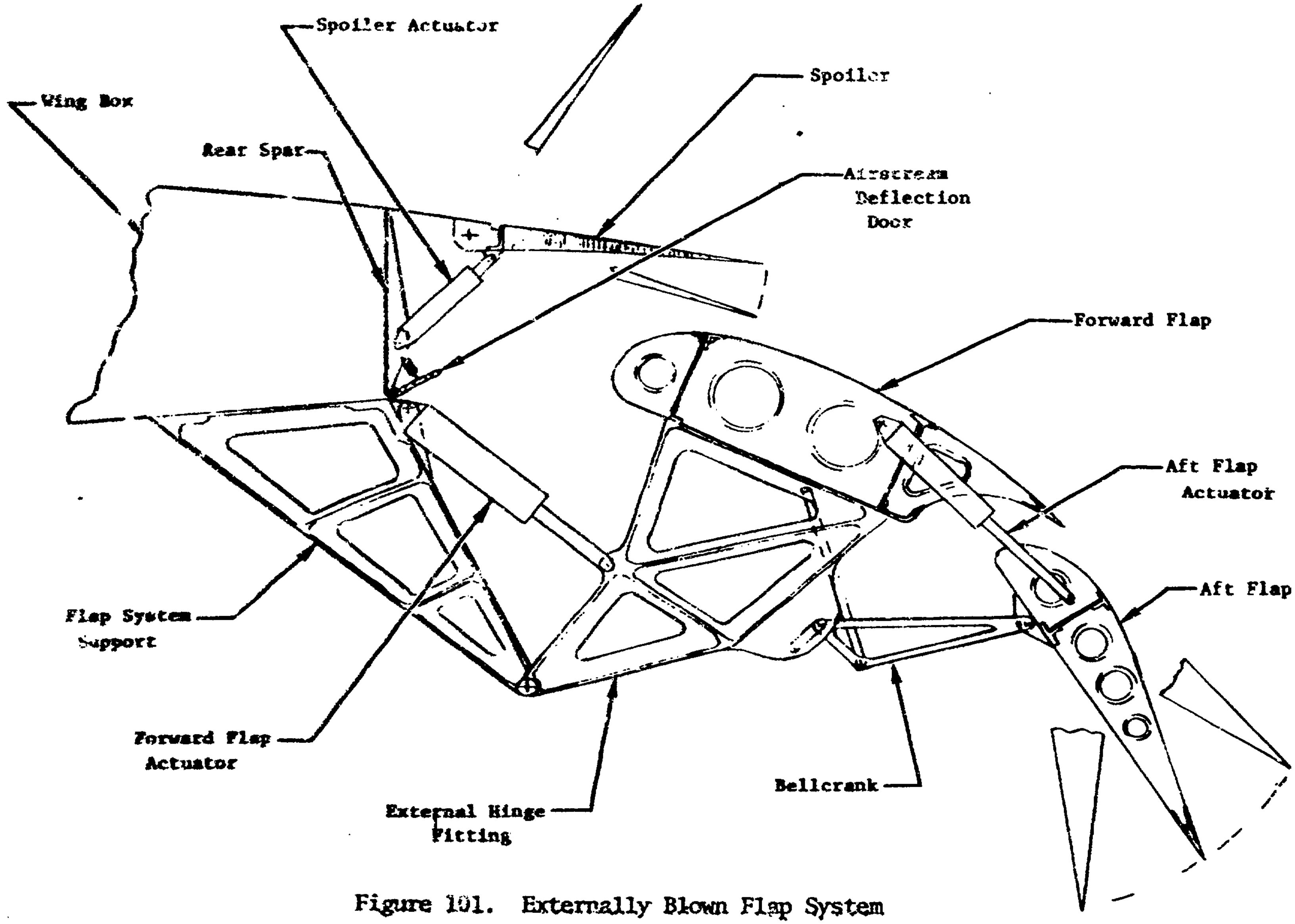


Figure 101. Externally Blown Flap System

- Lower maintenance costs
- Minimum weight
- Minimum construction costs
- Simple rigging of links
- Minimum field ground time

An external single hinge system was not chosen for the flaperon, since the increased depth at the wingtip is susceptible to damage by ground vehicle traffic.

The large two-part aileron system, illustrated in figure 102, is 36 percent of the wing chord and comprises the extreme outboard control surface segment. The forward panel extends to 25 degrees. The aileron pivots off the forward panel to 60 degrees with the added articulated capability of ±25 degrees. The aileron has independent control from the forward panel.

The material and construction for this system is of conventional 2024 and an experimental 7050 aluminum alloy which is used predominantly throughout with steel fastener-bearing combinations. The forward panel is of double spar and rib construction. The ribs are sized to take the high 134-decibel sonic pressure level of the ducted blown engine air. The links are simple die forged parts.

The aileron system is sized to the flutter criteria of 350,000 inch-pounds per radian.

No thermal criteria are imposed on these surfaces.

Each engine pylon-nacelle is supported off the wing with a box beam constructed pylon. The forward section tapers from a single-point forward engine mount to a rectangular shape over the rear engine mounts, and this rectangular-shaped beam further increases in size to the wing front spar and is mounted with vertical fittings. The lower part of the box beam continues aft of the front spar and fairs in to the lower wing surface. (See figure 103.)

The pylon structure box reacts the fore-and-aft front engine mount loads and the vertical and side loads and moments at the rear engine mounts; additional design consideration is given to engine thermal expansion. This box is a semimonocoque structure comprising upper and lower spars, bulkheads, and stiffened side panels. Titanium is used largely in this section to meet the combined effect of stiffness, thermal, and fire seal criteria.

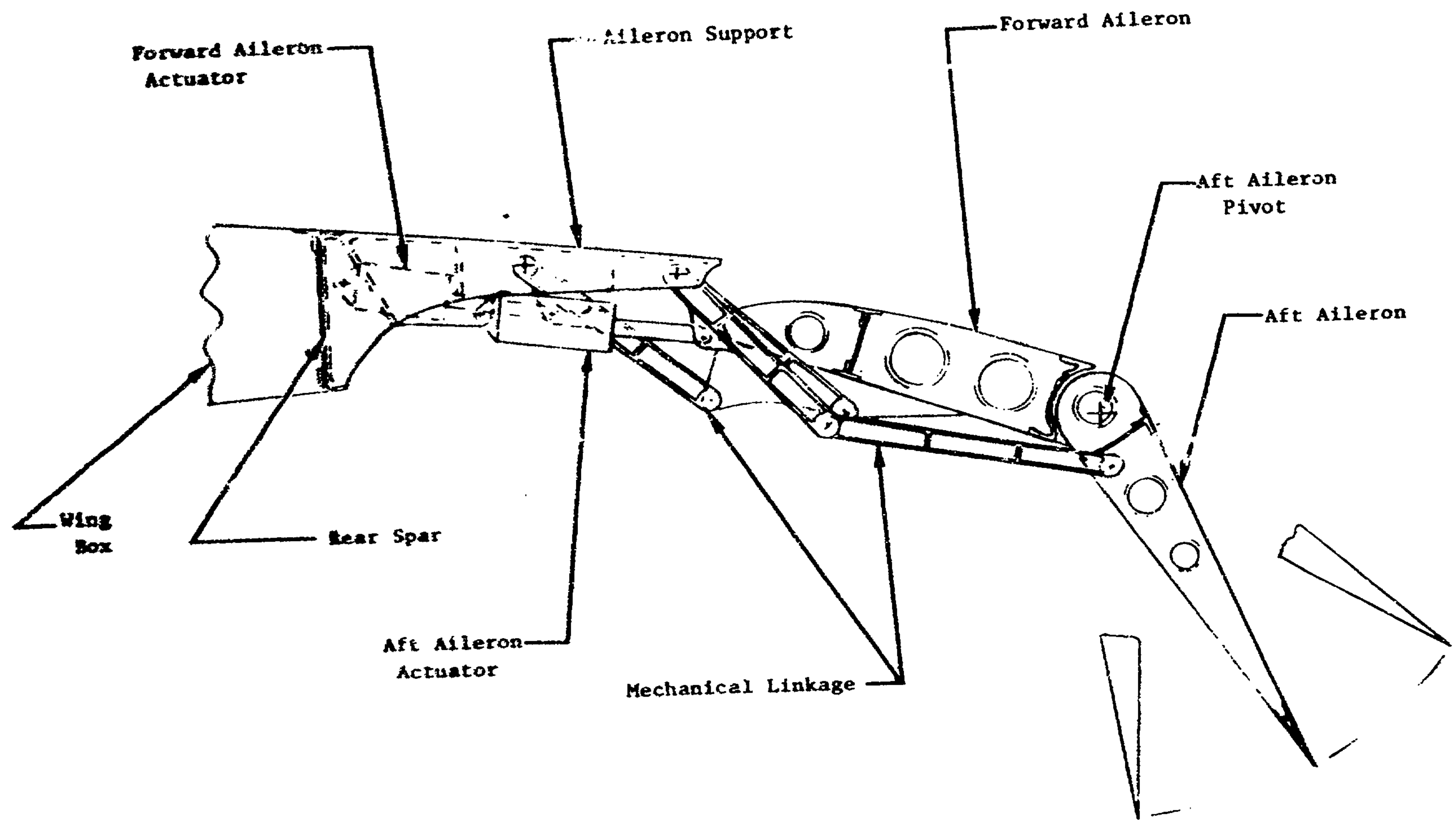


Figure 102. Two-Part Aileron System

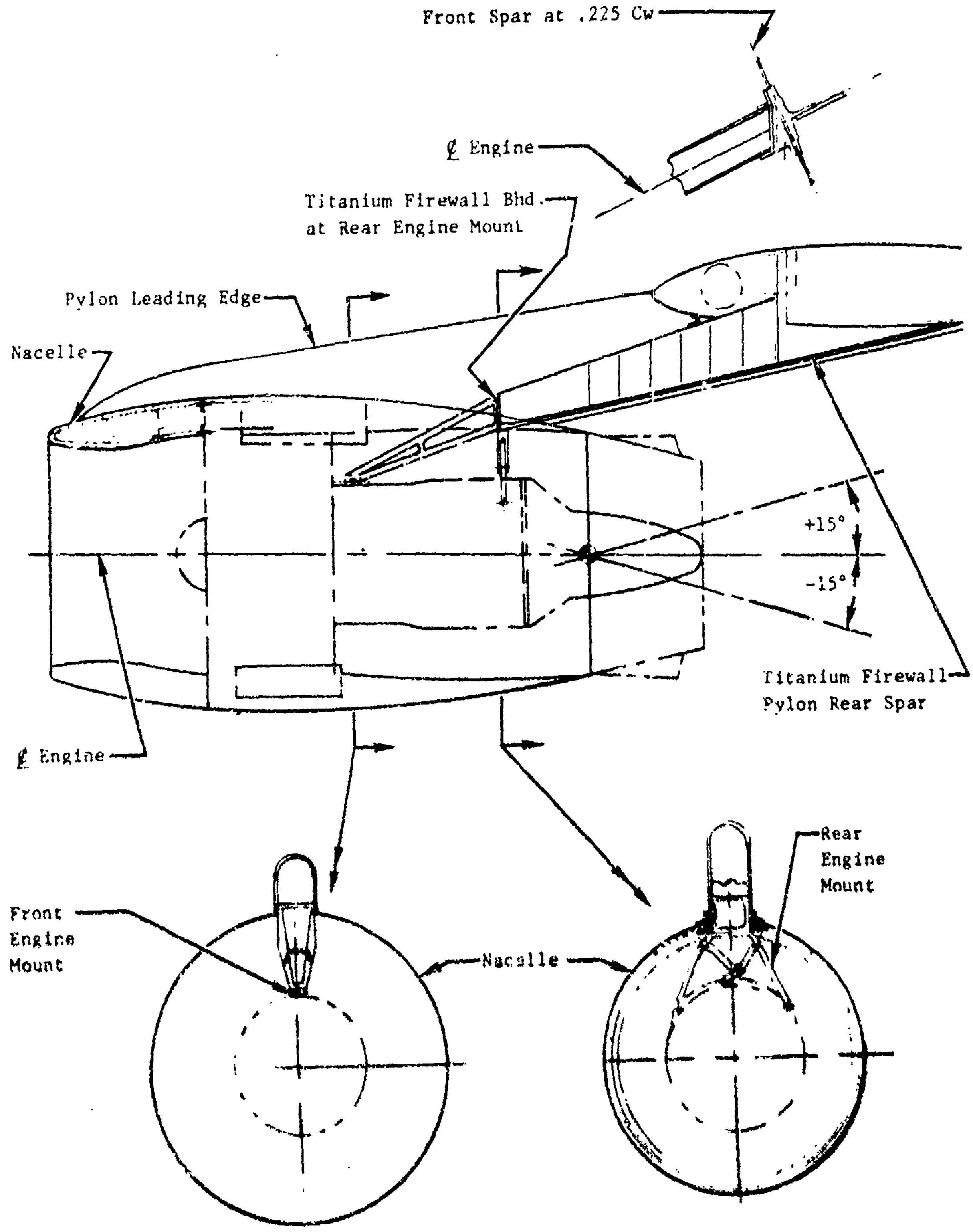


Figure 103. Pylon Structure

The pylon leading edge is of aluminum alloy stiffened construction and opens for systems access and inspection. The engine-fed air duct runs through this passage to the wing leading edge and out to the blown flaperons. The nacelle is circular in shape and is constructed largely of hinged honeycomb-stiffened panels to allow access throughout the engine. Nozzle loads are transmitted through the nozzle hinge fittings into the nacelle shell structure which is supported by the pylon.

8.2.3 EMPENNAGE

The empennage is a high T-tail configuration with double-hinged control surfaces on both the vertical and horizontal stabilizers as shown in figure 104. The horizontal and vertical stabilizers are similar in configuration and construction, and the following description applies equally to both surfaces, unless otherwise specified.

The aft control surface is hinged at the 75 percent chord line and deflects ±25 degrees by means of a linkage system composed primarily of bell-cranks, antifriction bearings, and push-pull rods connected to torque shafts in the control surfaces. This surface is used in a conventional manner for medium or high-speed control. The forward control surface is hinged at the 55-percent chord line, deflects ±25 degrees, and is connected through linkage to the aft control surface such that a 2:1 ratio in aft surface to forward surface deflection is produced. These fore-and-aft surfaces also operate independently, thereby allowing the forward system to be locked in neutral position while the aft system is activated.

A pitch actuator, mounted in the T-tail fairing, allows +10 to -20 degrees deflection of the horizontal stabilizer for longitudinal trim. This motion is independent of the fore-and-aft control surface deflections.

The fixed vertical and horizontal tail structure is of aluminum alloy material composed of stiffened machined skins, along with web-stiffened ribs in the outboard areas and truss-type ribs at the increasing depth inboard areas. The movable control surfaces are conventional formed skins, spars, and ribs.

The vertical tail spars continue through into the aft fuselage where the tail loads are reacted. (Refer to paragraph 8.2.1.)

8.3 STRUCTURE DATA

The data summarized represent the results of extensive and thorough parametric and definitive studies coupled with the use of the highly

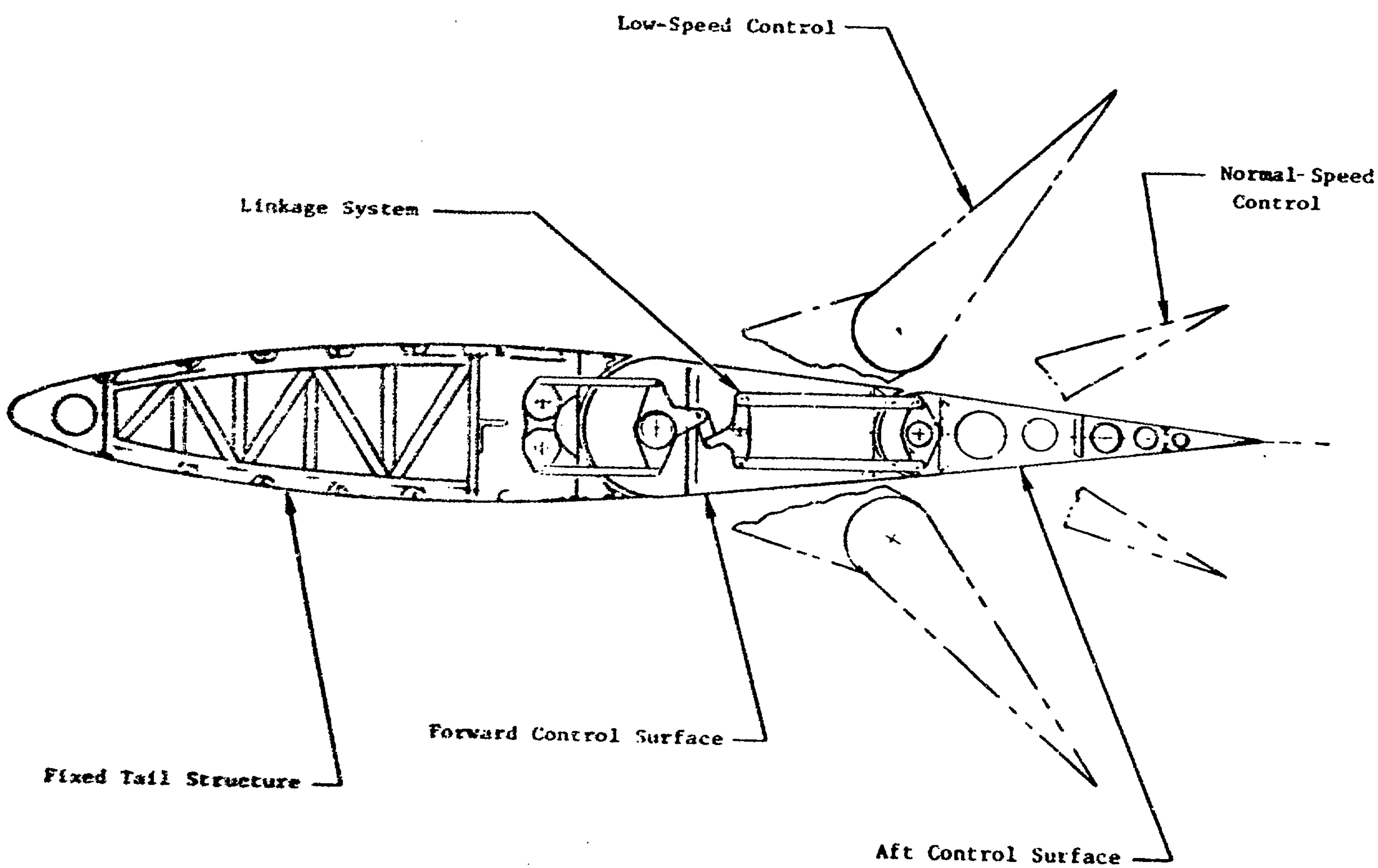


Figure 104. Typical Vertical and Horizontal Tail Structure

sophisticated structural weight estimation program (SWEEP). Studies were conducted on an in-house structural basepoint STOL airplane identified as D-516-2.

8.3.1 PRELIMINARY STRUCTURAL DESIGN CRITERIA

Table XV outlines the complete mission mixture of the D-516-2 preliminary baseline configuration.

Table XVI defines the airplane design gross weights used for structural design.

The design speeds (flaps up) are selected to be commensurate with the operational missions. The level flight maximum speed, V_H, is 335 knots equivalent airpseed (KEAS) or 0.75M, whichever is greater. That is, V_H is 335 KEAS from sea level to 20,000 feet, and at 20,000 feet and above V_H is 0.75M. The limit speed, V_L, is selected to be 1.1 times the level flight maximum speed, V_H. V_L is 370 KEAS from sea level to 20,000 feet and is equal to 0.825M at 20,000 feet and above. The variation of design speed versus altitude is shown in figure 105.

The design flap speed for a particular flap setting is 1.5 times the power-off stall speed for the next lesser flap setting or 1.3 times the power-on stall speed with the flap fully retracted, whichever is greater at design takeoff gross weight (TOGW).

The power-off stall at design TOGW is 133 KEAS, and the power-off staff speed at design TOGW with the flap setting of 20/40 is 105 [illegible]ing this rationale, the design flap speeds would be as follows:

V_F for δ_f = 20/40 is 1.5 x 133 = 200 KEAS

V_F for δ_f = 30/60 is 1.3 x 133 = 173 KEAS

NOTE 1.5 x 105 = 158 KEAS which is less than 173 KEAS.

At airplane weights up to the basic flight design weight (BFDW), the maximum limit maneuver load factors are +3.0 and -1.0. At the maximum overload design weight (MODW), the maximum limit maneuver load factors are +2.5 and -1.0. This envelope is shown in figure 106.

The limit maneuver load factors with the flaps extended are +2.0 and 0.0 for weights up to the maximum design landing weight of the BFDW. Figure 107 outlines the load factors for the extended flaps.

TABLE XV. MISSION MIX

Mission	Radius (n mi)	Payload (lb)	% Total Hr	Mission Time (hr)	Sorties	Flight Hours	Design T.O. Weight (lb)	Fuel Weight (lb)	Landings: Mode CTOL	Landings: Mode STOL	Landings: Total
I - Employment	500	28K	40	2.753	2,906	8,000.2	150,000	27,335	2,906	2,906	5,812
II - Employment	200	37.49K	12	1.338	1,794	2,400.4	150,000	17,845	1,794	1,794	3,588
IIIA - Deployment	3,700	0	2	9.283	43	399.2	155,000	60,000	43	0	43
IIIB - Deployment	2,600	34K	2	6.238	64	399.2	180,000	51,546	64	0	64
IIIC - Deployment	1,000	49.23K	2	2.538	157	398.5	171,800	25,900	157	0	157
IV - Low-altitude resupply		28K	30	1.500	4,000	6,000.0	150,000	27,335	4,000	12,000	16,000
V - Assault		28K	5	1.000	1,000	1,000.0	150,000	27,335	1,000	1,000	2,000
VI - Training		9.5K	7	2.405	584	1,404.5	150,000	45,830	2,351	2,351	4,702
Total			100		10,548	20,002.0			12,315	20,051	32,366

TABLE XVI. DESIGN GROSS WEIGHTS

Description	Structural Basepoint D-516-2	D-516-2A
Operating weight empty	94,665	98,149
Minimum flying weight (MNFW) **The operating weight empty (OWE) plus the weight of 5 percent of the total usable internal fuel**	97,665	101,350
Basic flight design weight (BFDW) **The gross weight at engine start with the primary design mission useful load - the 500 n mi radius employment mission with a 28,000 lb payload**	150,000	159,310

TABLE XVI. DESIGN GROSS WEIGHTS (CONCL)

Description	Structural Basepoint D-516-2	D-516-2A
Maximum overload design weight (MODW) The gross weight at engine start with the maximum useful load - the maximum of 34,000 lb plus the fuel required for a 2,600 n mi range plus the fuel required for an end of the mission 30-minute loiter at sea level plus a 5.0 percent fuel reserve	180,000	191,172
Landing design weight (LDW) The midmission gross weight for the primary design mission - the 500 n mi radius employment mission with a 28,000 lb payload	137,500	145,238
Maximum landing design weight (MLDW) The gross weight at engine start for the primary design mission - the same as the basic flight design weight	150,000	159,310

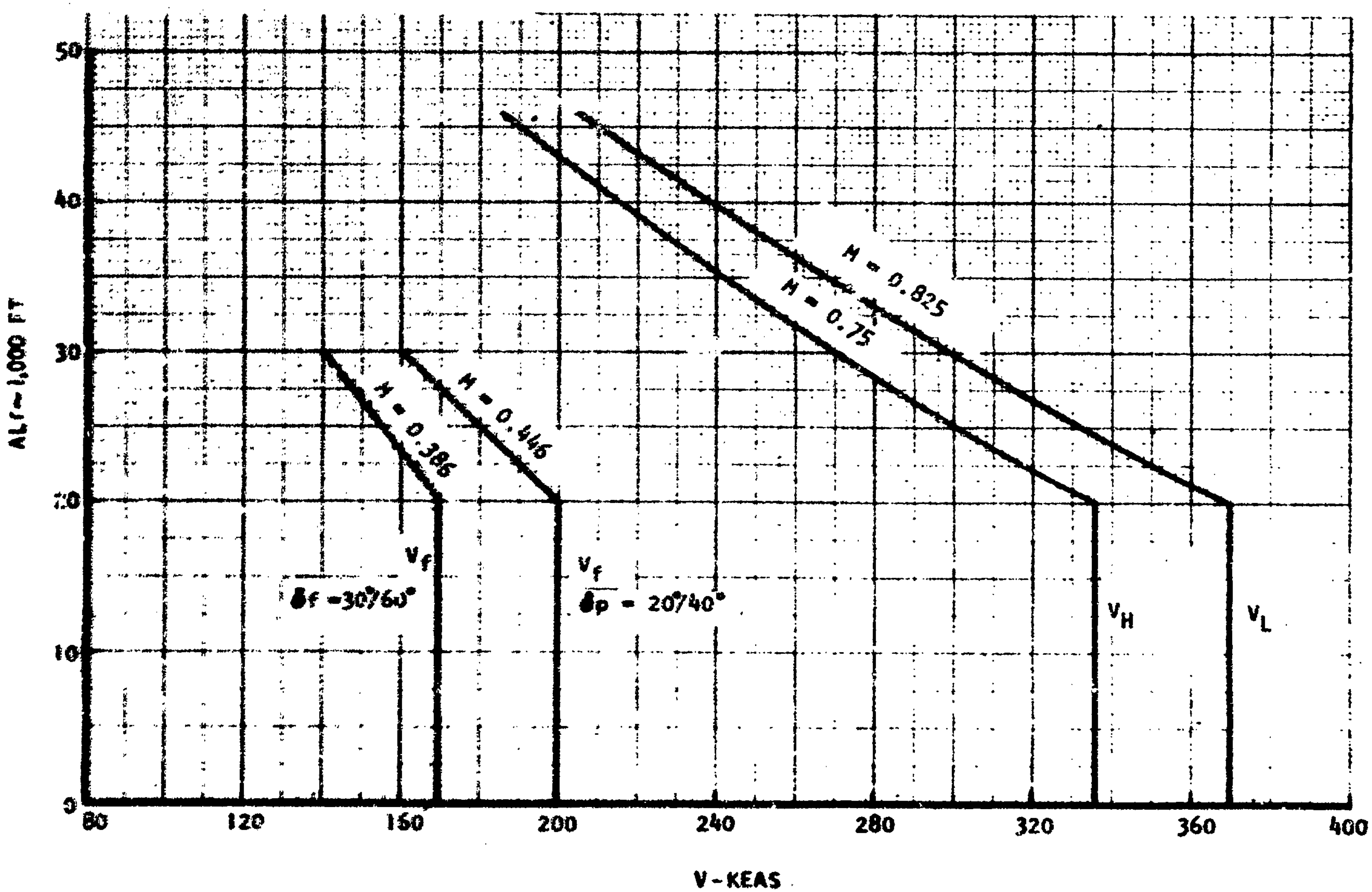


Figure 105. Lesign Speed Versus Altitude

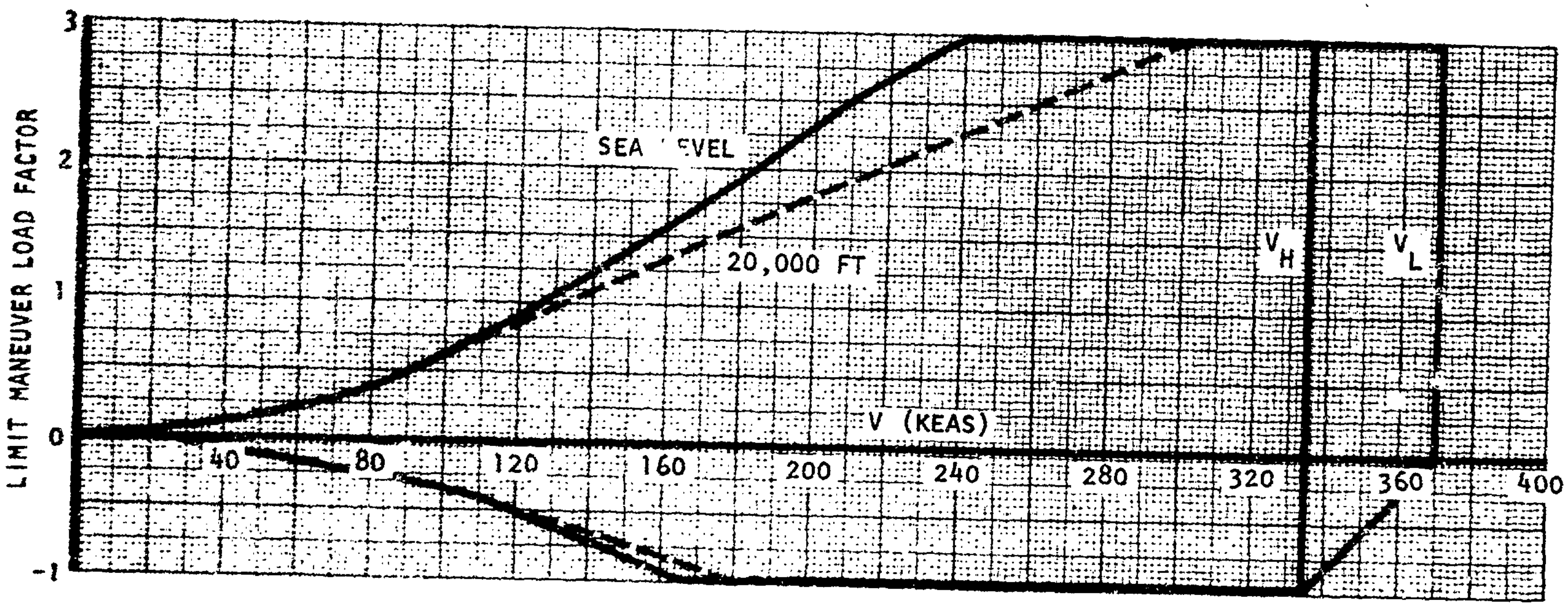


Figure 106. Maneuver Load Factor Envelope

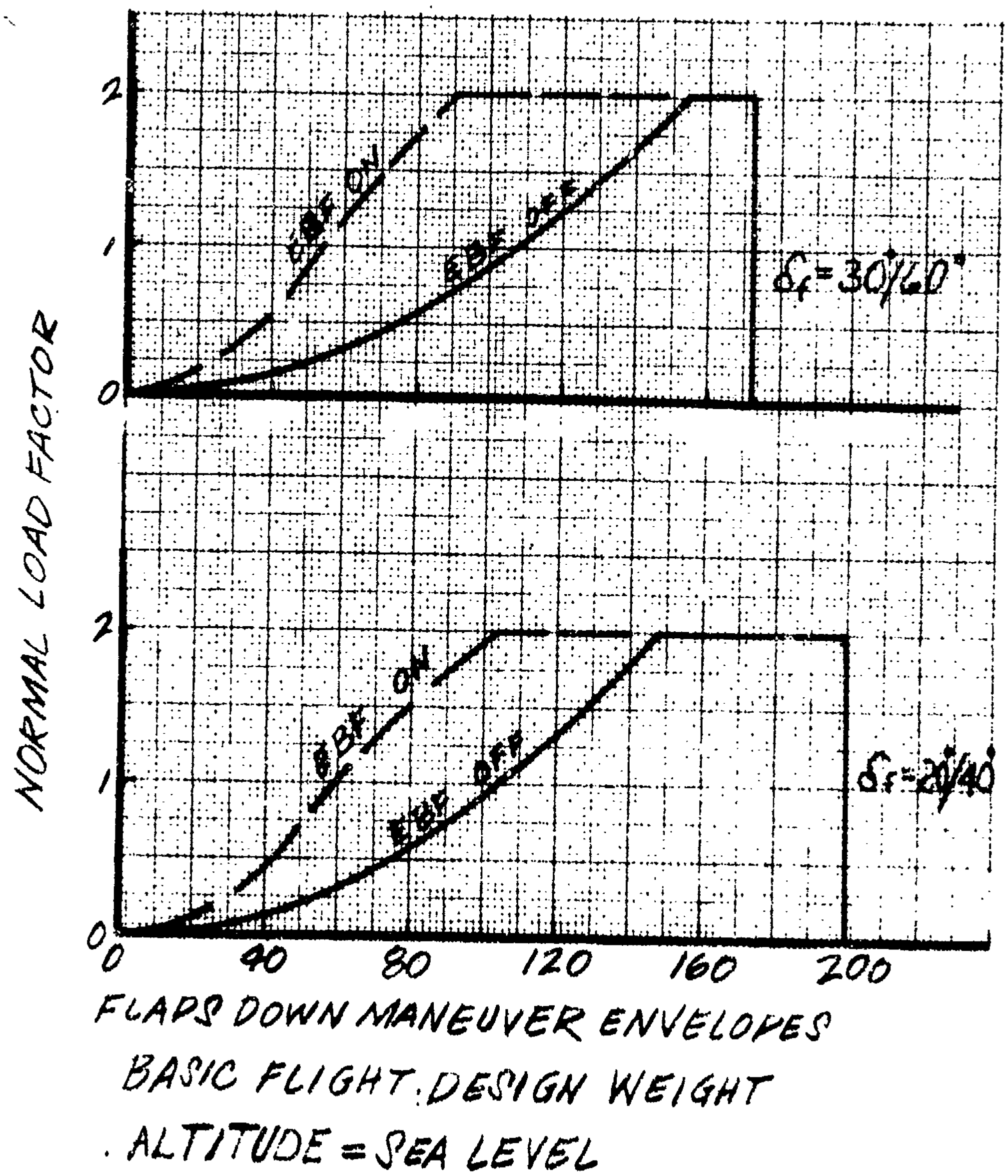


Figure 107. Flaps-Down Maneuver Envelope

The limit gust load factors are determined in accordance with the discrete gust analysis of paragraph 3.22.1 of MIL-A-00866A. At altitudes between sea level and 20,000 feet, the design equivalent gust velocities are 66 feet per second (fps) at V_G, 50 fps at V_H, and 25 fps at V_L. The speed V_G is defined at the speed where the 66 fps gust line intersects the $C_{N_{max}}$ line on the V-n diagram or equal to $V_S\sqrt{n_g}$, whichever is greater. The speed V_S in the flaps-up stall speed and n_g is the gust load factor for the 50 fps gust of V_H. The gust load factor envelope is shown in figure 108.

In addition, a gust rational probability analysis (RPA) was conducted to determine the wing root bending moment corresponding to an individual airplane failure probability of 0.0005 due only to atmospheric turbulence. The mission profiles and mission mix used are the same as used to develop the fatigue load spectra. The low-level contour turbulence RPA data are shown in figure 109.

The design limit landing load factors correspond to a design limit sink speed of 15 fps at the design landing weight.

8.3.2 STRUCTURE LOADS

Table XVII shows the airload conditions, and table XVIII shows the repeated load spectra conditions considered.

Shear, bending moment, and torsion loads on the airplane major structural components for all table XVII conditions were computed; the representative critical conditions for the structure components are summarized in figures 110, 111, and 112. Wing bending moment spectra were generated at two wing stations for all the conditions of table XVIII. The side-of-fuselage bending moment is at BP 72.0 and is about the unswept axis. Wing-outer station bending moment is at the wing swept station 271.84 and is about the swept axis. Scatter factors are not included in the bending moment spectra.

The net ultimate wing bending moments obtained from the RPA using an individual airplane failure probability of 0.0005 are as follows:

- At BP 72.0 M_x = 58,700,000 in.-lb ultimate
- At wing swept station 271.84 M_x = 25,700,000 in.-lb ultimate

The maximum net ultimate wing bending moments obtained from the discrete design conditions occur for condition 50717 and are as follows:

- At BP 72.0, $M_x = 1.5\ [46802480 - 3.0\ (4663000)]$

$$= 49{,}220{,}220 \text{ in.-lb ultimate}$$

- At wing swept station 271.84, $M_x = 1.5\ [21099344 - 3.0\ (1513000)]$

$$= 24{,}840{,}516 \text{ in.-lb ultimate}$$

Three critical limit load conditions on the wing flaps and ailerons are shown in table XIX with remaining loads being normal to the surface element chord plane; centers of pressure locations are aft of the surface element leading edge:

- Forward flap, $X_{cp} = 0.475\ C_{ff}$

- Aft flap, $X_{cp} = 0.350\ C_{af}$

A summary of maximum limit loads on various components of the structural basepoint aircraft, D-516-2, is given in table XX. This is the basepoint aircraft processed through the SWEEP computer system to arrive at the structure definition for the D-516-2A.

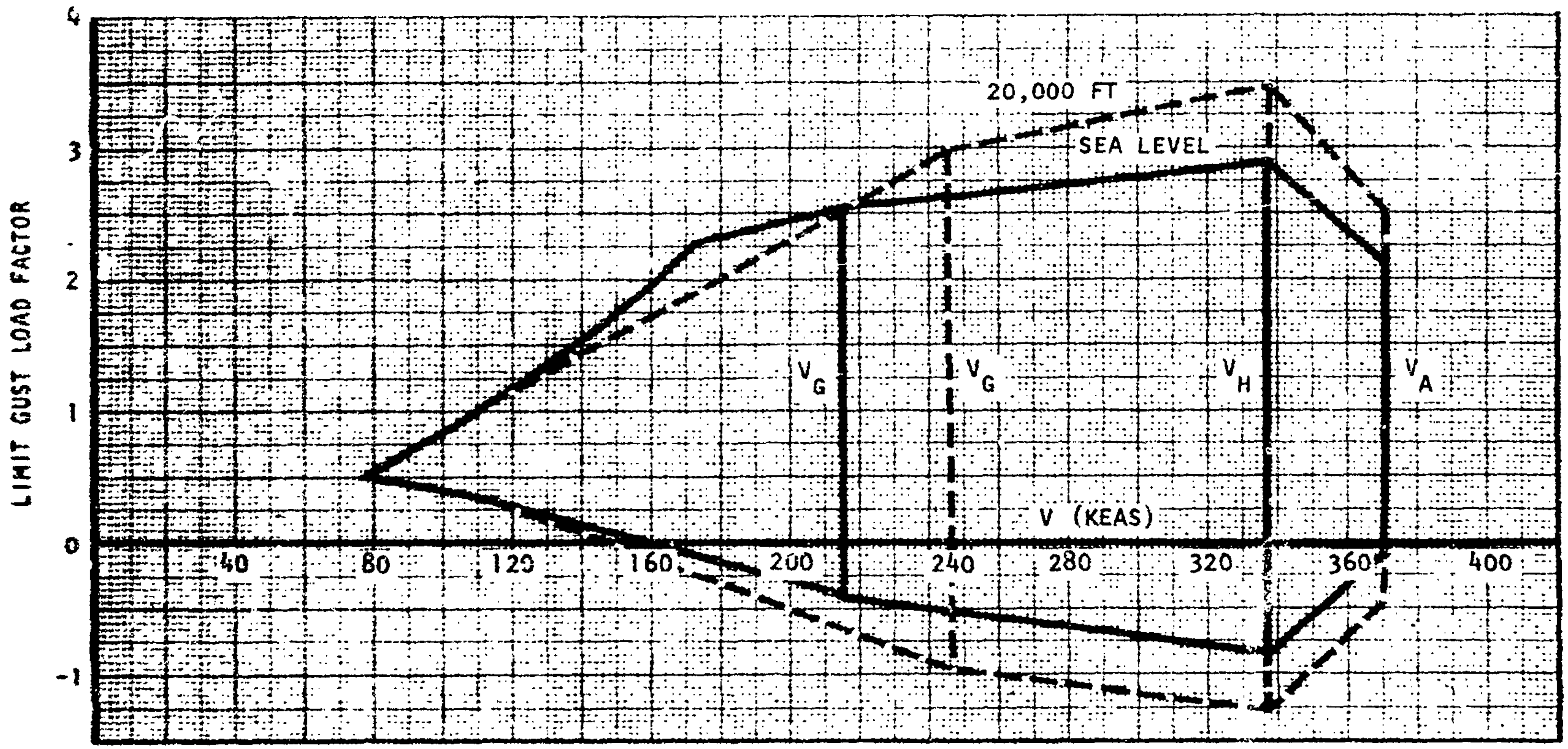


Figure 108. Gust Load Factor Envelope

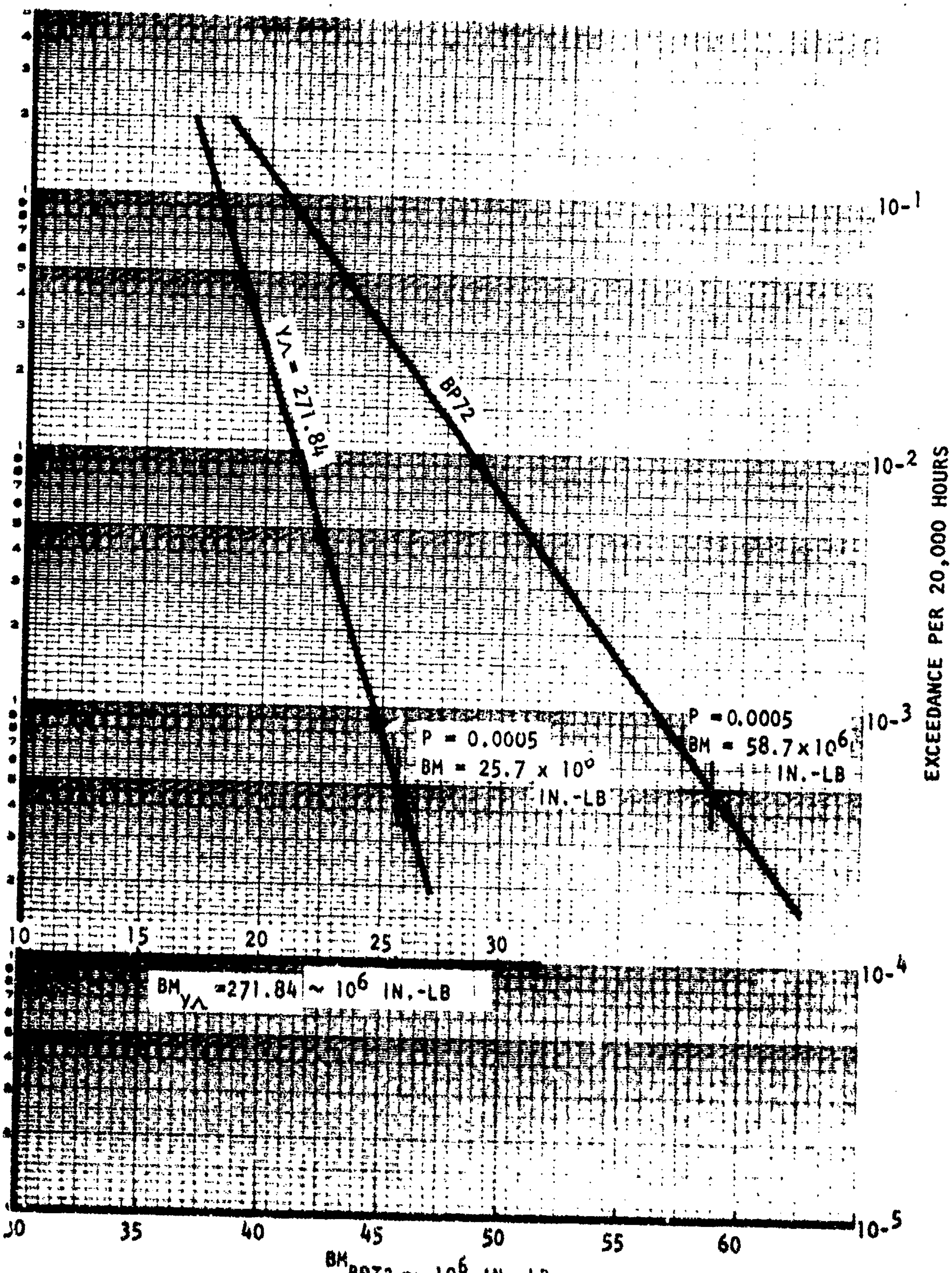


Figure 109. Rational Probability Analysis Bending Moment Using Low Level Turbulence

TABLE XVII. AIRLOAD CONDITIONS

Condition	Type
50117	+ 3.0 G maneuver, V_L at sea level
50217	+ 3.0 G maneuver, V_L at 20,000 ft
50617	- 1.0 G maneuver, V_H at sea level
50817	+ 2.0 G maneuver, flaps 30/60 at 173 KEAS
50917	1.0 G trim, flaps 30/60 at 125 KEAS
51017	+ Gust at V_H at sea level
51117	+ Gust at V_H at 20,000 ft
51417	- Gust at V_H at sea level
51517	- Gust at V_H at 20,000 ft
51817	Lateral gust at V_H at sea level
51917	Lateral gust at V_H at 20,000 ft
52017	Pitch acceleration at V_L at sea level
52117	Pitch acceleration at V_L at 30,000 ft
52217	Yaw acceleration at V_L at sea level
52317	Yaw acceleration at V_L at 30,000 ft

TABLE XVIII. REPEATED LOAD SPECTRA CONDITIONS

Segment	M	Alt (ft)	Weight (lb)	Time (hr)
1*	0.60	20,000	141,000	4,370
2*	0.60	20,000	165,000	88
3*	0.40	5,000	141,000	665
4*	0.75	40,000	141,000	6,682
5*	0.75	40,000	159,000	509
6*	0.65	20,000	141,000	708
7*	0.47	10,000	136,500	270
8*	0.456	1,000	136,500	6,708
9	Taxi spectrum at W = 150,000 lb			
10	Taxi spectrum at W = 137,400 lb			
11	GAG cycles for 32,366 flights			

*Type of condition = maneuver and gust

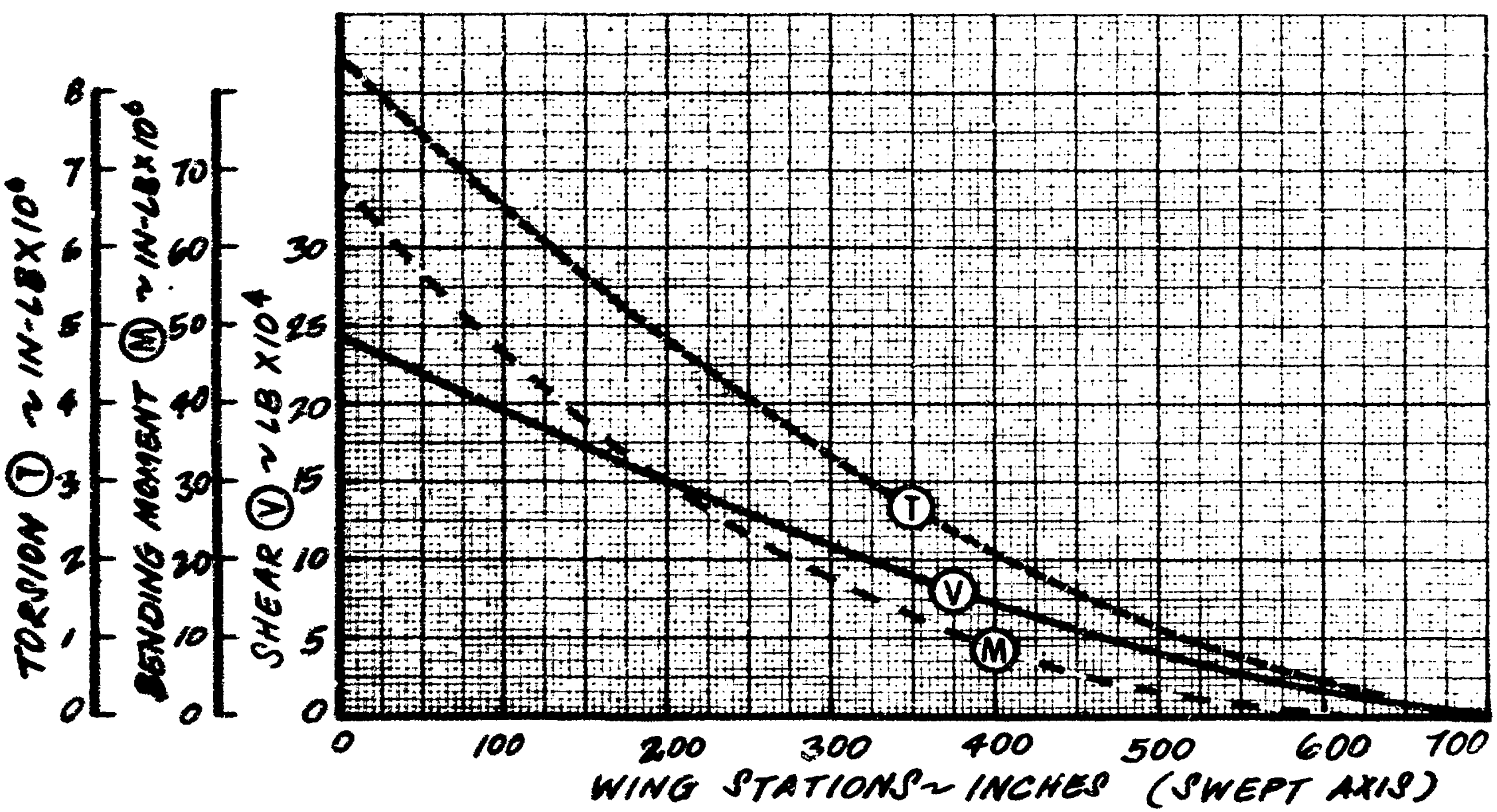


Figure 110. D-516-2A Wing Limit Airload - Condition 50217

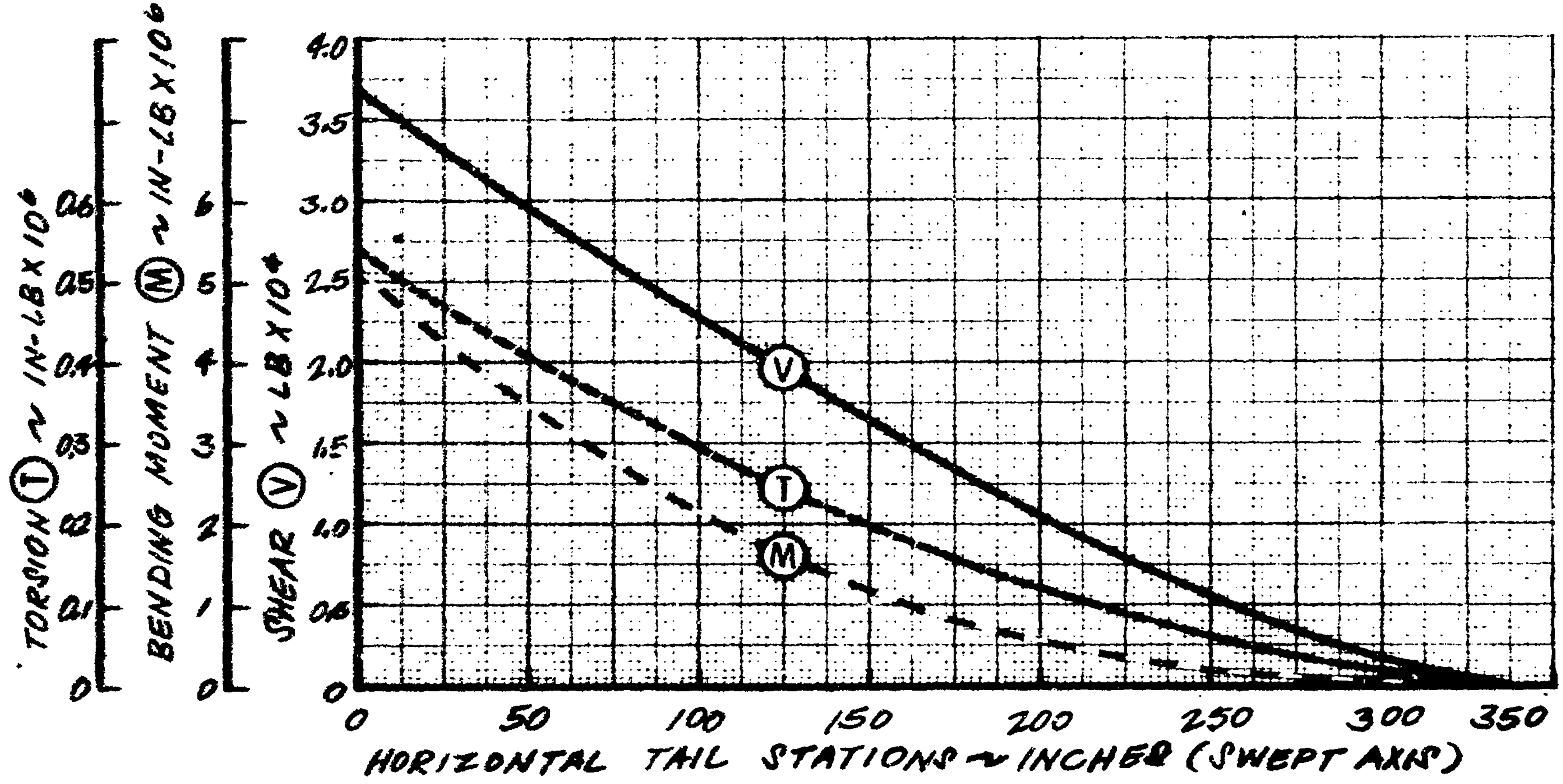


Figure 111. D-516-2A Horizontal Tail Limit Airload - Condition 51117

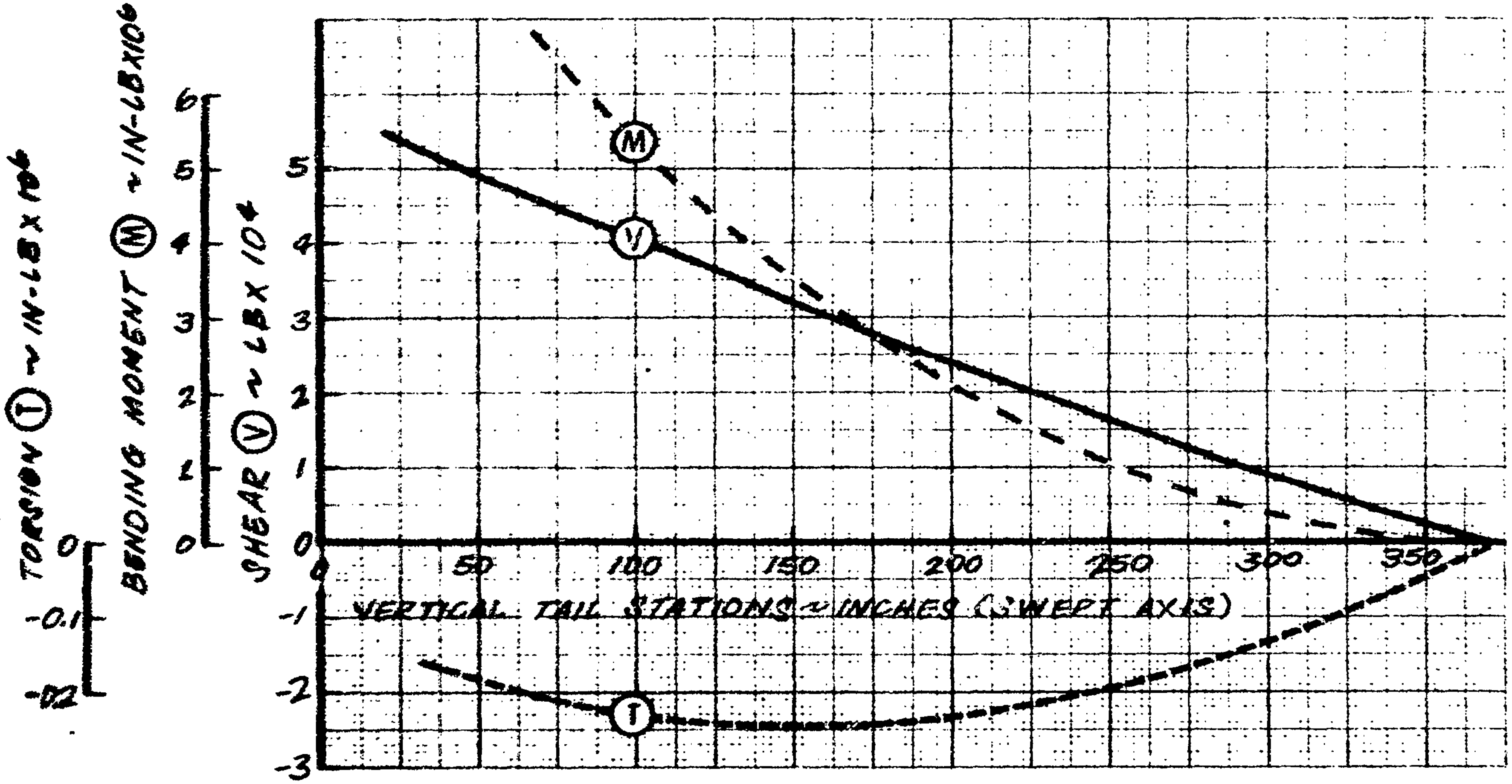


Figure 112. D-516-2A Vertical Tail Limit Airload - Condition 51917

TABLE XIX. WING FLAP AND AILERON LIMIT LOADS

Condition	Surface	Wing Sta BL (in.)	Running Load (lb per inch of span)	
			Fwd Flap	Aft Flap
V = 175 KEAS	Inbd flap	107	43.3	44.4
		240	41.6	32.8
δ_f = 20/40 (degrees)	Midflap	240	41.6	32.8
W = 150,000 lb		391	37.0	26.8
	Outbd flap	391	37.0	26.8
Power = max		516	24.5	20.6
N_Z = 2.0	Aileron	516	24.5	20.6
		586	20.4	17.6
		646	15.3	12.9
		688	8.2	7.0
V = 148 KEAS	Inbd flap	107	35.0	43.2
		240	31.9	24.0
δ_f = 30/60 (degrees)	Midflap	240	31.9	24.0
W = 138,000 lb		291	22.1	19.0
	Outbd flap	391	19.5	16.9
Power = max		516	16.3	15.1
N_Z = 2.0	Aileron	516	16.3	15.1
		586	13.3	12.5
		646	10.0	9.4
		688	5.4	5.0
V = 192 KEAS	Inbd flap	107	64.2	72.5
		240	65.3	54.0
δ_f = 30/60 (degrees)	Midflap	240	65.3	54.0
W = 150,000 lb		391	66.8	49.0
	Outbd flap	391	66.8	49.0
Power = off		516	53.8	42.8
N_Z = 2.0	Aileron	516	53.8	42.6
		586	44.2	35.6
		646	33.4	20.0
		688	18.1	14.3

TABLE XX. SUMMARY OF MAXIMUM LIMIT LOADS ON COMPONENTS - D-516-2 AIRPLANE

Component	Condition	Gross Weight (10^3 lb)	Cargo Weight (10^3 lb)	Speed	Altitude (10^3 ft)	Load Factor	Shear (10^3 lb)	Component Load BM (10^6 in.-lb)	Component Load Tors (10^6 in.-lb)
Wing root (BP 72)	Pullup	150	28	0.825M	20	3.0	182.4	46.8	-14.9
Wing station 272*	Pullup	150	28	0.825M	20	3.0	100.8	17.1	2.34
Horizontal tail root	Vertical gust	150	28	0.75M	20	3.38	41.2	5.44	-1.88
Vertical tail root	Lateral gust	150	28	0.75M	20	nz = 1.0 ny = 1.48	53.9	7.50	-5.30
Flaps	Flap 30°/60° pullup	150	28	173 KEAS	0	2.0	Fwd flap 54 lb/in. of span Aft flap 58 lb/in. of span		
Flaps	Flaps 20/40° pullup	150	28	200 KEAS	0	2.0	Fwd flap 59 lb/in. of span Aft flap 55 lb/in. of span		
Slats at wing root	Pullup	150	28	200 KEAS	0	2.0	Normal load 142 lb/in. Chord load -59 lb/in.		
Slats at wingtip	Pullup	150	28	200 KEAS	0	2.0	Normal load 42 lb/in. Chord load -18 lb/in.		
Cargo floor	Vertical gust	150		0.75M	20	3.38	1,000 psf		

*Wing station 310 along swept reference line at 0.465 C_W

Section IX

MASS PROPERTIES

9.1 WEIGHT, BALANCE, AND INERTIA

The weight analyses of the D516-2A configuration were accomplished by using the integrated Structures Weight Estimating Program, "SWEEP," and other weight-estimating methods developed at NR. The "SWEEP" was developed under an ASD contract, and is an integration of Wings, Fuselage, and Landing Gear Groups estimating programs.

The fixed equipment and Auxiliary Power System Group weights were recommended by customer and used as is. Approximate STOL penalties are included.

A weight summary is given in table XXI and the group weight statement follows for the D516-2A configuration.

The balance and inertia data shown in table XXII were developed through the use of the balance and inertia program. The configuration was analyzed once after the preliminary sizing and again for the final vehicle.

The airplane configuration balance and inertia calculations were made with the deliverable payload located at the center of the cargo bay. The center of gravity is within the balance limits.

9.2 ACCURACY ANALYSIS

The accuracy analysis data were developed from representative sample aircrafts. The pertinent parameters were compiled for each sample airplane and an empirical weight estimating equation was derived using the parameters for each "group weight items," e.g., wing, body, propulsion, etc.

The empirical weight-estimating equations were developed by the NR regression and correlation analysis program. The "best fit" equation, in linear, exponential, or logarithmic form, was used. The resulting equations are listed in table XXIII and the pertinent parameters are included in table XXIV. The symbols and definitions of these parameters are shown in table XXV.

A summary of the D516-2A group weights and the operating weight empty (OWE) is shown in table XXVI. The actual weight column refers to the estimated weights, based on current NR weight-estimating methods. The estimated weight column reflects those group weight items estimated by the empirical equation as previously discussed.

The accuracy analysis was performed on the OWE and based on four sample cargo airplanes. The resultant data are shown in figure 113. The OWE prediction range for 80 and 90 percent confidence interval was based on the following:

$$\text{range} = \pm\, t_{\left(\frac{\alpha}{2}, v\right)} S_{Y \cdot X} \sqrt{\frac{1}{n} + \frac{(X^{*} - \bar{X})^2}{\sum_{i=1}^{n} (X_i - X)^2}}$$

Where range = upper and lower limits of mean OWE

t = student's t distribution

α = level of significance, subscripted

v = degree of freedom, subscripted

$S_{Y \cdot X}$ = sample standard deviation

$$S_{Y \cdot X} = \sqrt{\sum_{i=1}^{n} (Y_i - \bar{Y})^2 - \frac{\left[\sum_{i=1}^{n} (X_i - \bar{X})(Y_i - \bar{Y})\right]^2}{\sum_{i=1}^{n} (X_i - \bar{X})^2} \quad \frac{1}{(n-2)}}$$

n = number of samples

X^{*} = assumed OWE

X_i = estimated OWE of samples

$\bar{X}$ = mean estimated OWE of samples

$$= \sum_{i=1}^{n} \frac{(X_i)}{n}$$

Y_i = actual OWE of sample

$\bar{Y}$ = mean actual OWE of sample

Sample	Y_i	X_i
C-141	131320	126108
KC-135	95138	92282
C-130	68492	68995
C-133	116653	117860

Mean actual OWE $\bar{Y}$ = 102900.75

Mean estimated OWE $\bar{X}$ = 101311.25

$$\sum_{i=1}^{n} Y_i = 411603$$

$$\sum_{i=1}^{n} (Y_i)^2 = 44595251917$$

$$\sum_{i=1}^{n} (X_i) = 405245$$

$$\sum_{i=1}^{n} (X_i)^2 = 430/0484810$$

$$\sum_{i=1}^{n} (X_i\ Y_i) = 43814355600$$

$$S_{Y \cdot X} = 3315.726542$$

Confidence interval $_{(1)}$ = 100 (1 - α) = 90 percent

Confidence interval $_{(2)}$ = 100 (1 - α) = 80 percent

Where α = 0.10 for 90 percent

α = 0.20 for 80 percent

Student's "t" distribution (reference 1)

For $\alpha = 0.10,\ t\left(\frac{\alpha}{2}, v\right) = 2.92$

$\alpha = 0.20,\ t\left(\frac{\alpha}{2}, v\right) = 1.886$

Range Evaluation

Assumed OWE (X*)	Range (±)	
	90%	80%
94959	5051	3249
100,000	4849	3132
130,000	7851	5075
70,000	8310	5367

The D516-2A estimated versus actual OWE falls outside of the +90 percent confidence interval. The weight difference of approximately 10,000 pounds can be attributed to the following factors:

1. 4.5g ultimate load factor at basic flight design gross weight

2. Externally blown flap airloads, temperature, and acoustic penalties

3. High sink rate (15 fps) for landing weight defined as midmission weight

4. CBR of 6 and 200 passes for the landing gear

5. Double-hinged elevator and rudder control surfaces

6. Double aileron panels for high- and low-speed operation

7. Fatigue penalties added to landing gear structures

8. Addition of ground mobility provision

9. Droop horizontal leading edge

10. Airplane growth factor

TABLE XXI. WEIGHT SUMMARY

Item	Weight (lb)
• Total Structure Group	(65,420)
Wing	18,980
Horizontal tail	3,210
Vertical tail	2,940
Body	24,570
Main landing gear	7,345
Nose landing gear	1,080
Surface control	2,885
Nacelle section	4,410
• Propulsion Group	(17,350)
Engine(s)	14,445
Fuel system	2,365
Engine controls	125
Starting system	415
• Auxiliary Power System	(500)
• Fixed Equipment Group	(12,100)
Instruments	900
Hydraulics	900
Electrical	1,900
Electronics	2,000
Armor	700
Furnishings	(4,000)
Personnel accommodation	
Miscellaneous equipment	
Furnishings	
Emergency equipment	
Cargo handling	
Air-conditioning and anti-ice	1,600
Auxiliary gear	100
• Total - Weight Empty	(95,370)
• Ground Mobility	(87[illegible])
Total	(96,240)

TABLE XXI. WEIGHT SUMMARY (CONCL)

Item	Weight (lb)
• Operating Weight Empty (OWE) Items	(1,909)
Crew	860
Oil, engine	200
Liquid nitrogen	185
Unusable fuel	664
• Operating Weight Empty	(98,149)
Weight empty	96,240
OWE items	1,909
• Initial Takeoff Gross Weight	(159,310)
Operating weight empty	98,149
Payload, deliverable	28,000
Fuel, initial	33,161

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GROUP WEIGHT STATEMENT
WEIGHT EMPTY

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1	WING GROUP					18980
2	CENTER SECTION - BASIC STRUCTURE				2201	
3	INTERMEDIATE PANEL - BASIC STRUCTURE					
4	OUTER PANEL - BASIC STRUCTURE (INCL. TIPS LBS.)				9446	
5						
6	SECONDARY STRUCTURE (INCL. WINGFOLD MECHANISM LBS.)				1177	
7	AILERONS (INCL. BALANCE WEIGHT LBS.)				436	
8	FLAPS - TRAILING EDGE				3251	
9	- LEADING EDGE				1746	
10	SLATS					
11	SPOILERS				723	
12	SPEED BRAKES					
13						
14						
15	TAIL GROUP					6150
16	STABILIZER - BASIC STRUCTURE				2017	
17	FINS - BASIC STRUCTURE (INCL. DORSAL LBS.)				2044	
18	SECONDARY STRUCTURE (STAB. & FINS)				654	
19	ELEVATOR (INCL. BALANCE WEIGHT LBS.)				780	
20	RUDDERS (INCL. BALANCE WEIGHT LBS.)				655	
21						
22						
23	BODY GROUP					24570
24	FUSELAGE OR HULL - BASIC STRUCTURE				14854	
25	BOOMS - BASIC STRUCTURE					
26	SECONDARY STRUCTURE - FUSELAGE OR HULL				922	
27	- BOOMS					
28	- SPEEDBRAKES					
29	- DOORS, PANELS & MISC.				7154	
30	RESTRAINT, RAILS, CONVEYORS, ETC.				1640	
31	ALIGHTING GEAR GROUP - LAND (TYPE:)					8425
32–33	LOCATION	WHEELS, BRAKES TIRES, TUBES, AIR	STRUCTURE	CONTROLS		
34	MAIN	2754	3771	820	7345	
35	AUX	513	406	161	1080	
36						
37						
38						
39						
40	ALIGHTING GEAR GROUP - WATER					
41	LOCATION	FLOATS	STRUTS	CONTROLS		
42						
43						
44						
45						
46	SURFACE CONTROLS GROUP					2885
47	COCKPIT CONTROLS				160	
48	AUTOMATIC PILOT				335	
49	SYSTEM CONTROLS (INCL. POWER & FEEL CONTROLS LBS.)				2390	
50						
51	ENGINE SECTION OR NACELLE GROUP					4410
52	INBOARD					
53	CENTER					
54	OUTBOARD					
55	DOORS, PANELS & MISC.					
56						
57	TOTAL (TO BE BROUGHT FORWARD)					65420

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GROUP WEIGHT STATEMENT
WEIGHT EMPTY

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		Auxiliary	Main	
1	PROPULSION GROUP			17350
2		AUXILIARY	MAIN	
3	ENGINE INSTALLATION		14445	
4	AFTERBURNERS (IF FURN. SEPARATELY)			
5	ACCESSORY GEAR BOXES & DRIVES			
6	SUPERCHARGERS (FOR TURBO TYPES)			
7	AIR INDUCTION SYSTEM			
8	EXHAUST SYSTEM			
9	COOLING SYSTEM			
10	LUBRICATING SYSTEM			
11	TANKS			
12	COOLING INSTALLATION			
13	DUCTS, PLUMBING, ETC.			
14	FUEL SYSTEM		2365	
15	TANKS - PROTECTED			
16	- UNPROTECTED			
17	PLUMBING, ETC.			
18	WATER INJECTION SYSTEM			
19	ENGINE CONTROLS		125	
20	STARTING SYSTEM		415	
21	PROPELLER INSTALLATION			
22				
23				
24	AUXILIARY POWER PLANT GROUP			500
25	INSTRUMENTS & NAVIGATIONAL EQUIPMENT GROUP			900
26	HYDRAULIC & PNEUMATIC GROUP			900
27				
28				
29	ELECTRICAL GROUP			1900
30				
31				
32	ELECTRONICS GROUP			2000
33	EQUIPMENT			
34	INSTALLATION			
35				
36	ARMAMENT GROUP (INCL. GUNFIRE PROTECTION LBS.)			700
37	FURNISHINGS & EQUIPMENT GROUP			4000
38	ACCOMMODATIONS FOR PERSONNEL			
39	MISCELLANEOUS EQUIPMENT			
40	FURNISHINGS			
41	EMERGENCY EQUIPMENT			
42				
43	AIR CONDITIONING & ANTI-ICING EQUIPMENT GROUP			1600
44	AIR CONDITIONING			
45	ANTI-ICING			
46				
47	PHOTOGRAPHIC GROUP			
48	AUXILIARY GEAR GROUP			100
49	HANDLING GEAR			
50	ARRESTING GEAR			
51	CATAPULTING GEAR			
52	ATO GEAR			
53	GROUND MOBILITY			870
54				
55	MANUFACTURING VARIATION			
56	TOTAL FROM PG. 2			65420
57	WEIGHT EMPTY			96240

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GROUP WEIGHT STATEMENT
USEFUL LOAD & GROSS WEIGHT

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1	LOAD CONDITION				BASIC MISSION	EXTENDED RANGE		
2								
3	CREW (NO. 4)				860	860		
4	PASSENGERS (NO.)							
5	FUEL	Type	Gals.					
6	UNUSABLE	JP-4	102		664	664		
7	INTERNAL	JP-4	5102		33161			
8	INTERNAL	JP-4	9845			63992		
9								
10	EXTERNAL							
11								
12.	BOMB BAY							
13								
14	OIL				200	200		
15	TRAPPED							
16	ENGINE							
17								
18	FUEL TANKS (LOCATION)							
19	WATER INJECTION FLUID (GALS)							
20								
21	BAGGAGE							
22	CARGO - DELIVERABLE				28000	29031		
23								
24	ARMAMENT							
25	GUNS (Location)	Fix. or Flex.	Qty.	Cal.				
26								
27								
28								
29								
30								
31								
32	AMMUNITION							
33								
34								
35								
36								
37								
38								
39	INSTALLATIONS (BOMB, TORPEDO, ROCKET, ETC.)							
*40	BOMB OR TORPEDO RACKS							
41								
42								
43								
44								
45								
46	EQUIPMENT							
47	PYROTECHNICS							
48	PHOTOGRAPHIC							
49								
*50	OXYGEN							
51	LIQUID NITROGEN				185	185		
52	MISCELLANEOUS							
53								
54								
55	USEFUL LOAD				63070	94932		
56	WEIGHT EMPTY				96240	96240		
57	GROSS WEIGHT				159310	191172		

*If not specified as weight empty.

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GROUP WEIGHT STATEMENT
DIMENSIONAL & STRUCTURAL DATA

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		Main Floats	Aux. Floats	Booms	Fuse or Hull	Nacelles Inboard	Nacelles Center	Nacelles Outboard
1	LENGTH - OVERALL (FT.)	103.08 Ft.		HEIGHT - OVERALL - STATIC (FT.)		44.8		
2								
3	LENGTH - MAX. (FT.)				103.08	16.25		16.25
4	DEPTH - MAX. (FT.)				16.74	7.33		7.33
5	WIDTH - MAX. (FT.)				17.5	7.33		7.33
6	WETTED AREA (SQ. FT.)				5081	320.0		320.0
*7	FLOAT OR HULL DISPL. - MAX (LBS.)							
8	FUSELAGE VOLUME (CU. FT.)			PRESSURIZED	16046	TOTAL	18399	

		Wing	H. Tail	V. Tail
9				
10	GROSS AREA (SQ. FT.)	1600	665	410
11	WEIGHT/GROSS AREA (LBS./SQ. FT.)	11.86	4.827	7.17
12	SPAN (FT.)	113.13	54.7	23.96
13	FOLDED SPAN (FT.)			
14				
15	SWEEPBACK - AT 25% CHORD LINE (DEGREES)	25	25	30
16	- AT % CHORD LINE (DEGREES)			
**17	THEORETICAL ROOT CHORD - LENGTH (INCHES)	280	208.40	256.70
18	- MAX. THICKNESS (INCHES)	35	25	30.8
***19	CHORD AT PLANFORM BREAK - LENGTH (INCHES)			
20	- MAX. THICKNESS (INCHES)			
***21	THEORETICAL TIP CHORD - LENGTH (INCHES)	86.68	83.36	154.02
22	- MAX. THICKNESS (INCHES)	8.67	6.67	15.4
23	DORSAL AREA, INCLUDED IN (FUSE.) (HULL) (V. TAIL) AREA (SQ. FT.)			
24	TAIL LENGTH - 25% MAC WING TO 25% MAC H. TAIL (FT.)	54.92		

25	AREAS (SQ. FT.) Flaps	L.E.	T.E. 309		
26	Lateral Controls	Slots 210	Spoilers 197	Ailerons 69	
27	Speed Brakes	Wing	Fuse. or Hull		
28					
29					

		NOSE		MAIN	
30	ALIGHTING GEAR (LOCATION)	NOSE		MAIN	
31	LENGTH - OLEO EXTENDED - ℄ AXLE TO ℄ TRUNNION (INCHES)				
32	OLEO TRAVEL - FULL EXTENDED TO FULL COLLAPSED (INCHES)	25"		30"	
33	FLOAT OR SKI STRUT LENGTH (INCHES)				
34	ARRESTING HOOK LENGTH - ℄ HOOK TRUNNION TO ℄ HOOK POINT (INCHES)				
35	HYDRAULIC SYSTEM CAPACITY (GALS.)				

		Location	No. Tanks	****Gals. Protected	No. Tanks	****Gals. Unprotected
36	FUEL & LUBE SYSTEMS					
37	Fuel - Internal	Wing	2	3826		6019
38		Fuse. or Hull				
39	- External					
40	- Bomb Bay					
41						
42	Oil - ENGINE	NACELLE			4	28.3
43						
44						

		Fuel in Wings (Lbs.)	Gross Weight	Ult. L.F.
45	STRUCTURAL DATA - CONDITION	Fuel in Wings (Lbs.)	Gross Weight	Ult. L.F.
46	FLIGHT	33161	159310	4.5
47	LANDING	19089	145238	
48				
49	MAX. GROSS WEIGHT WITH ZERO WING FUEL			
50	CATAPULTING			
51	MIN. FLYING WEIGHT	3200	101349	
52	LIMIT AIRPLANE LANDING SINKING SPEED (FT./SEC.)		15	
53	WING LIFT ASSUMED FOR LANDING DESIGN CONDITION (%W)		100	
54	STALL SPEED - LANDING CONFIGURATION - POWER OFF (KNOTS)			
55	PRESSURIZED CABIN - ULT. DESIGN PRESSURE DIFFERENTIAL - FLIGHT (P.S.I.)			13.2
56				
57	AIRFRAME WEIGHT (AS DEFINED IN AN-W-11) (LBS.)			74952

*Lbs. of sea water @ 64 lbs./cu. ft.
**Parallel to ℄ or ℄ airplane.
***Parallel to ℄ airplane.
****Total usable capacity.

CONTRACT

MODEL D516-2A

REPORT NO.

NR SERIAL NO.

DATE

CONTRACT	
MODEL	
WEIGHT EMPTY	95570
Less:	
Engine	14445
Trapped Oil and Coolant	
Propeller, Power Control, and Governor	
Wheels, Brakes, Tires, Tubes, and Air	3267
Auxiliary Power Plant	242
Turbo Superchargers	
Radio Receivers, Transmitters, Radar, and Removable Units	1430
Starter	120
Battery	34
Generator and/or Alternator	373
Turrets and Power Operated Gun Mounts	
Electronics and Fire Control System	
Instruments	416
Air Conditioning and Pressurization Units	300
Rubber or Nylon Fuel Cells	661
AIRFRAME WEIGHT (AMPR)	74082
TAKE-OFF GROSS WEIGHT	159310
FLIGHT DESIGN GROSS WEIGHT	145238
*STRUCTURAL WEIGHT	65420
WING AREA (SQUARE FEET)	1600

*Structural Weight includes Weight of Surface Controls to agree with AN 9103D

TABLE XXII. BALANCE AND INERTIA

Balance - Lemac = 450 MAC = 189.00

Condition	Gross Weight (lb)	X	Z	Percent Mac
Operating weight empty	98,149	515.73	215.00	34.78
OWE + 28 KPL	126,149	510.02	205.04	31.76
OWE + fuel	131,310	504.18	227.13	28.67
TOGW gear down	159,310	501.69	217.44	27.35
TOGW gear up	159,310	498.74	219.24	25.79
Midmission gear down	145,240	504.76	213.03	28.97
Midmission gear up	145,240	501.52	215.00	27.26

Inertia - Weight in lb, inertia in slug-ft^2

Condition	Gross Weight (lb)	X	Z	Pitch I_y	Roll I_x	Yaw I_z	Product I_{xz}
OWE	98,149	515.73	215.01	1664073	1143546	2367523	320491
OWE + 28 KPL	126,149	510.02	205.46	1833181	1173191	2527988	325714
OWE + fuel	131,310	504.18	227.13	1722752	1342162	2598703	308783
TOGW gear down	159,310	501.69	217.44	1896055	1378159	2726878	312675
TOGW gear up	159,310	498.74	219.24	1886662	1363404	2762240	322817

TABLE XXIII. WEIGHT ESTIMATING EQUATIONS

- **Wing**

$$\text{Index} = \frac{S_W^{0.725}\ AR^{0.436}\ (nG)^{0.516}\ (10\lambda)^{0.127}}{(100\ t/c)^{0.186}\ (100\cos\Lambda_{0.25})^{0.686}}$$

$$\text{Est} = e^{-0.8375883}\ \text{Index}^{1.063501}$$

- **Body**

$$\text{Index} = \frac{(0.05)\ S_F^{\ 1.124}\ (nG)^{0.172}\ Q^{0.241}}{(L_F/D_F)^{0.065}\ L_T^{\ 0.047}}$$

$$\text{Est} = e^{0.2763744}\ \text{Index}^{0.9747381}$$

- **V-Tail**

$$\text{Index} = \frac{(0.85)\ (nG/1000)^{0.376}\ Q^{0.122}\ S_V^{\ 0.875}\left(AR/\cos^2\Lambda_{0.25}\right)^{0.357}\ (10\lambda)^{0.039}}{(100\ t/c/\cos\Lambda_{0.25})^{0.489}}$$

$$\text{Est} = e^{1.299007}\ \text{Index}^{0.8387955}$$

TABLE XXIII. WEIGHT ESTIMATING EQUATIONS (CONT)

- H-Tail

$$\text{Index} = \frac{(0.26)\ (nG/1000)^{0.414}\ Q^{0.168}\ \left(AR/\cos^2 \Lambda_{0.25}\right)^{0.043}\ (10\lambda)^{0.025}\ S_H^{0.896}}{\left(100\ t/c \big/ \cos \Lambda_{0.25}\right)^{0.121}}$$

$$\text{Est} = e^{-7.239301}\ \text{Index}^{1.90465[illegible]}$$

- Surface Control

$$\text{Index} = (nG \times 10^{-4})^{0.514}$$

$$\text{Est} = e^{3.410074}\ \text{Index}^{1.847886}$$

- Landing Gear

$$\text{Index} = (0.0049\ S_S + 0.957)\ (0.04135\ L_M^{1.372} + 1.053)\ (1.86\ W_L^{0.6425} - 183.5)$$

$$\text{Est} = e^{-4.469931}\ \text{Index}^{1.542919}$$

- Engine Section

$$\text{Index} = (W_{eng})^{1.483}\ (10)^{-3}$$

$$\text{Est} = e^{3.636784}\ \text{Index}^{0.6091741}$$

TABLE XXIII. WEIGHT ESTIMATING EQUATIONS (CONT)

- Instrument Section

$$\text{Index} = (W_{I\ \text{equip.}})^{0.156}$$

$$\text{Est} = e^{0.8995342}\ \text{Index}^{6.157858}$$

- Hydraulics Section

$$\text{Index} = (N_{SS})^{1.496}\ (L_F)^{0.128}\ (D_E)^{0.3?2}$$

$$\text{Est} = e^{1.838292}\ \text{Index}^{1.007090}$$

- Electrical Section

$$\text{Index} = 0.0795\ (K_W)\ (L_E)\ (10)^{-2}$$

$$\text{Est} = e^{7.427040}\ \text{Index}^{0.1397161}$$

- Electronics Section

$$\text{Index} = W_{E\ \text{equip.}}$$

$$\text{Est} = e^{9.352112}\ \text{Index}^{-0.2358217}$$

TABLE XXIII. WEIGHT ESTIMATING EQUATIONS (CONCL)

- Furnishings Section

$$\text{Index} = (W_{UL})^{0.175}$$

$$\text{Est} = e^{4.275665}\ \text{Index}^{1.996657}$$

- ECS

$$\text{Index} = (W_{UL})^{0.33}$$

$$\text{Est} = e^{4.018951}\ \text{Index}^{1.047421}$$

- Propulsion Section

$$\text{Index} = (W_{eng})^{0.704}$$

$$\text{Est} = e^{3.417155}\ \text{Index}^{0.9541690}$$

TABLE XXIV. INDEXING VARIABLES

Variables	C-141	KC-135	C-123	C-130	C-133	C-140A	D516-2A
DGW	316100	275000	54000	135000	275000	40911	159310
N_{ULT}	3.75	3.0	4.5	4.5	3.75	4.5	4.5
Q	416	480	187	326	255	325	383
S_W	3228	2433	1223	1475	2673	542.5	1600
AR_W	8.0	7.03	9.9	10.07	12.07	5.3	8.0
λ_W	0.334	0.331	0.53	0.52	0.228	0.338	0.31
$(t/c)_W$	0.13	0.166	0.17	0.18	0.17	0.12	0.120
$\Lambda_{0.25}$	25°	35°	0°	0°	0°	30°	25°
S_H	483	500	345.5	545	801		665
AR_H	5.24	3.14	4.5	5.09	4.5		4.5
λ_H	0.370	0.447	0.349	0.391	0.349		0.40
$(t/c)_H$	0.105	0.11	0.12	0.108	0.12		0.112
$\Lambda_{0.25}$	25°	35°	9.9°	8.5°	9.9°		25°
S_V	416	312	186.7	300	640		410
AR_V	1.24	1.47	1.65	1.78	1.549		1.40
λ_V	0.609	0.35	0.432	0.296	0.344		0.60
$(t/c)_V$	0.13	0.11	0.116	0.15	0.12		0.116
$\Lambda_{0.25}$	35°	31°	14°	18.2°	12.2°		30

TABLE XXIV. INDEXING VARIABLES (CONCL)

Variables	C-141	KC-135	C-123	C-130	C-133	C-140A	D516-2A
S_F	5645	4420	2250	3460	6472		5081
L_T	74.5	60.9	40.7	43.6	66.3		54.9
L_F	132.29	128.83	76.25	97.75	153.37		103.08
D_F	14.16	13.83	11.67	13.25	16.1		16.96
D_E	38.3	45.4		36.0	40.7		27.67
W_L	257500	150000	51350	117732	245000		145240
S_{SP}	10.0	9.0	9.8	9.0	9.0		15.0
L_M	5.14	7.66	6.16	4.8	5.11		5.11
K_W	202.5	121.0		140.0	120.0		175
L_E	93.5	82.6		49.5	108.8		92.0
W_{UL}	38439	24844		35444	118411		29909
N_{SS}	11	7		7	5		11
W_E equip.	1281	1437		1431	1191		1430
W_{eng}	20720	14860		10648	15868		14445
W_I equip.	543	363	160	330	255		416

TABLE XXV. DEFINITION OF VARIABLES

S_W	Wing area
AR	Aspect ratio of wing
Λ	Sweep angle of wing
N_{ULT}	Ultimate load factor
G	Gross weight
λ	Taper ratio of wing
t/c	Thickness ratio - wing
S_F	Fuselage wetted area
L_T	Tail arm
L	Length of fuselage
D	Maximum diameter of fuselage
S_V	Vertical tail area
AR_V	Aspect ratio of vertical tail
Λ_V	Sweep angle of vertical tail
λ_V	Taper ratio of vertical tail
$(t/c)_V$	Thickness ratio - vertical tail
S_H	Horizontal tail area
AR_h	Aspect ratio of horizontal tail
Λ_H	Sweep angle of horizontal tail
λ_H	Taper ratio of horizontal tail
$(t/c)_H$	Thickness ratio - horizontal tail
S_S	Sink speed (feet/second)
W_L	Landing weight
L_M	Length of main gear (feet)

TABLE XXV. DEFINITION OF VARIABLES (CONCL)

W_{eng}	Weight of engines (installed)
N_{SS}	Number of hydraulic sybsystems
L_F	Length of fuselage
D_E	Distance from fuselage centerline to outboard engine
$W_{I\ equip.}$	Weight of instrument equipment
K_W	Kilowatts
L_E	Distance from forward to aft electrical distribution panels
$W_{E\ equip.}$	Weight of electronics equipment
W_{UL}	Maximum useful load less fuel weight

TABLE XXVI. OPERATING WEIGHT EMPTY - GROUP WEIGHT COMPARISON FOR D516-2A CONFIGURATION

Item	Actual (lb)	Estimated (lb)
Total Structure Group	65,420	54,989
Propulsion Group	17,850	18,977
Fixed Equipment Group	12,100	11,330
Useful Load	1,909	1,909
Total OWE	[98,149]	[87,205]

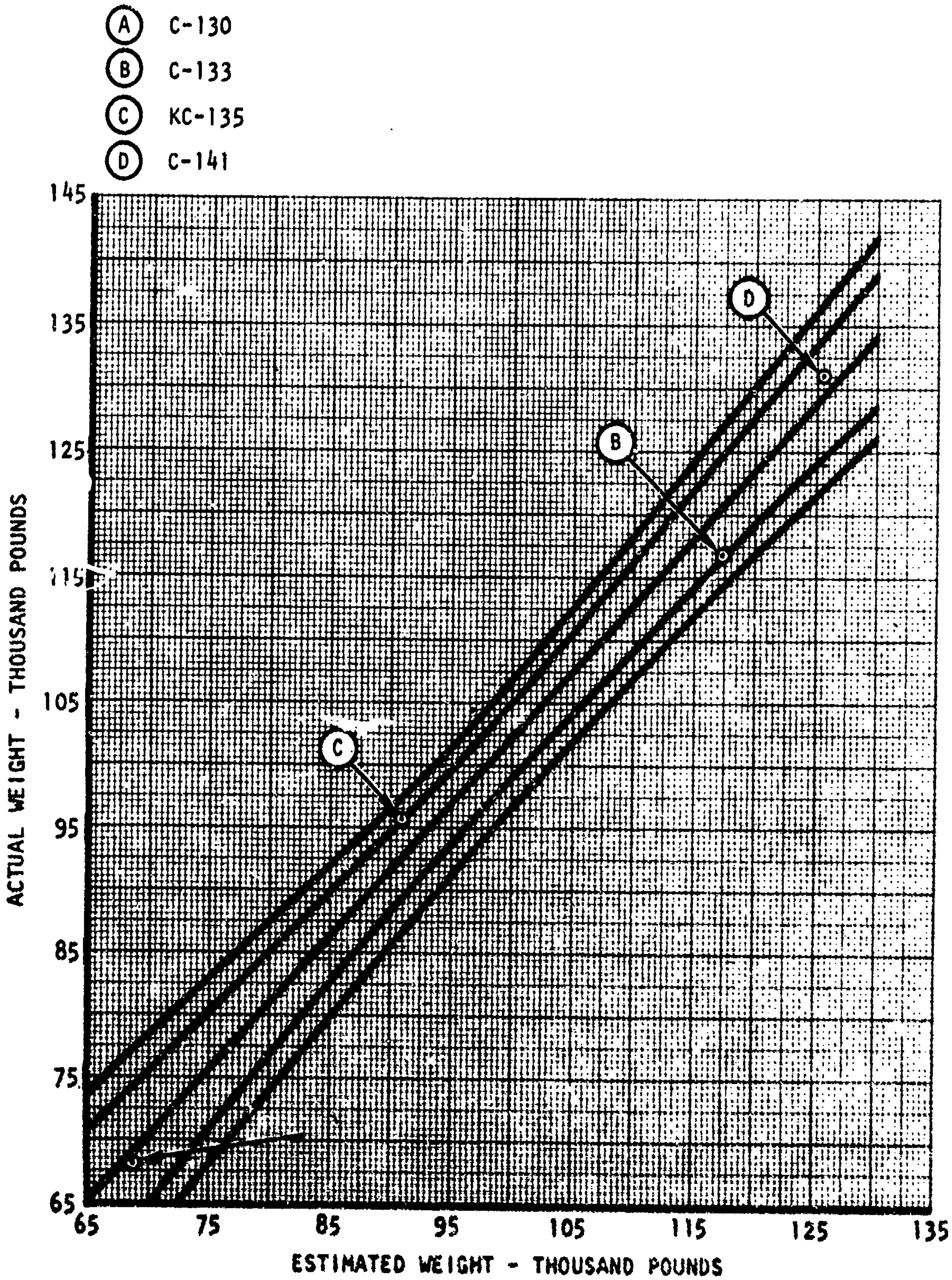

Figure 113. Accuracy of Estimation of 80- and 90-Percent Confidence OWE Regression and Cargo Aircraft

TABLE XXVII. COMPARISON OF OWE, ACTUAL VERSUS ESTIMATED

	Actual OWE	Estimated OWE	Ratio (X)
C-141	131,320	128,890.94	1.0188456
KC-135	95,138	93,335.187	1.0193148
C-130	68,492	69,036.500	0.99211282
C-133	112,495	120,340.19	0.934794
			$\frac{\Sigma X}{n} = \frac{3.9650672}{4}$
			$= 0.99127048$

9.3 METHODS AND DATA

The weight analysis encompasses two configuration sizing cycles, including the preliminary configuration and the resizing exercise following configuration refinement. In addition, a wing and empennage matrix was generated for the configuration sizing program.

The primary method of analysis is analytical, with utilization of other estimating means where applicable. The final takeoff gross weight and fuel weight were generally fixed prior to the final weight analysis. The final weight statement includes adjustment to individual pieces. The STOL or unique feature penalties are listed; however, these penalties are difficult to ascertain as no direct comparison could be made. The items listed are approximations of the "obvious" additions such as the blown flap system. A fatigue penalty was added to the landing gear system weight. No formal analysis was made to substantiate the penalty number.

9.3.1 WING GROUP

The estimated wing weight was derived using the following weight prediction methods:

1. A design-oriented structural analysis weight-estimating program.
2. Statistical estimating methods and data that were adjusted to reflect the STOL transport requirements
3. Preliminary structural sizing and design layouts

The design requirements and assumptions for the STOL transport wings are shown in table XXVIII. Material description is given in table XXIX. These data plus preliminary design drawings form the basis of design parameters used to analyze the wing torque box and control surfaces. The wing construction is basically a three-segmented, skin-stringer type with 7050-T7351 aluminum. The three segmented areas are: (1) the outboard section (swept trailing edge), (2) unswept trailing edge section to the side of fuselage, and (3) the center section.

The selection of 7050-T7351 aluminum as the primary structural material over 2024-T851 aluminum is estimated to reduce the wing torque box cover weights by approximately 450 pounds - 105 pounds in the upper cover and 345 pounds in the lower cover. This weight reduction is due principally to the superior strength, fatigue, and fracture toughness characteristics of the 7050 aluminum alloys.

9.3.1.1 Wing Analysis

Each segment of the wing was analyzed at 11 section cuts across the assumed structural axis, where the geometry, loads, stiffness, types of constructions, and design constraints are defined.

The section geometry was computed from local cross-section depths which were computed based on a normalized airfoil, local chord lengths, and thickness ratios. The cross sections were idealized into a rectangular box of width equivalent to the length of the structural chord between front and rear spars and the average depth.

At each wing box section, the net limit loads were defined in terms of vertical shear and spanwise bending moments. The moments were resolved into an average ultimate axial load (N_X) for the cover analysis. This approach does not allow for the forward or aft section of the box being designed by different loading conditions. The effects of different loading conditions and also of the MC/I effect were assumed to be accounted for by the indexing coefficients shown in table XXX.

The torsional rigidity requirements to prevent flutter were assumed to have negligible effects on the torque box. This is based on evaluation of a analogous wing, but with a higher velocity. A small flutter penalty appeared on the analogous wing, in the outboard section. However, with an optimized distribution, the flutter penalty can be eliminated, therefore, the flutter requirements are not a factor in the STOL wing analysis.

TABLE XXVIII. DESIGN CRITERIA AND ASSUMPTION - WING

Item	Value
Design gross weight (pounds)	159,310
Loading ($+N_Z/-N_Z$)	3.0/1.0
Fatigue cutoff (percent ftu)	85
Q-flutter (psf)	383
Material	7050-T7351 aluminum
Torque box construction	Multirib/sheet stringer
FS location	15 percent
RS location	63 percent
EA location	39 percent

TABLE XXIX. MATERIAL DESCRIPTION

Material	7050-T7351 Aluminum
E	10.5×10^6 psi
G	3.9×10^6 psi
F_{TU}	76,000 psi
F_{SU}	42,200 psi
F_{BRU}	97,000 psi
F_{CY}	66,000 psi
F_{PL}	52,800 psi
ρ	0.102 pound per cubic inch
μ	0.33

TABLE XXX. PROGRAM CALIBRATION AND NONOPTIMUM INDEXING FACTORS FOR WING TORQUE-BOXES

Torque-Box Element	Incremental Factors
Upper cover, average total	+0.2380
Lower cover, average total	+0.2690
Local misc shim pads, runouts, etc, average upper and lower	+0.0325
Ribs and bulkheads	+0.7500
Front spar, average total	+0.9500
Rear spar, average total	+1.1500
Misc and attachments, average	
Total	+0.1000

Special increments for functional design requirements for wing box are:

1. Front spar includes +0.6150 to account for leading edge slats effects and provisions.
2. Rear spar includes +0.950 to account for trailing edge flaps, ailerons and spoiler effects, and provisions.

Program estimates are incremented to allow for additional incremental weights - spanwise splices for upper and lower covers, pylon attach provisions, cutout provisions for access doors, fuel probes, drains, etc, and an average delta torque box weight of +6-1/2 percent is included for these effects.

9.3.2 WING WEIGHT ANALYSIS

The structural wing weight analysis was performed from data developed by a structural synthesis and weight analysis program. This program computes cover and substructure components weights for various types of torque box construction and materials. The analysis is controlled so that the synthesized structure closely approximates the structural criteria specified for the box and the arrangement of nonoptimum structures defined by the wing requirements and by standard design practices. The design constraint data, such as requirements for ribs, spar and rib spacings, types of stringer stiffener configurations, and locations of joints, are part of the input data to the program. Thus, the synthesized structures contain some nonoptimum design features. The cover, support, shear material requirements are synthesized and described in terms of geometry, thickness, and volumes. The description at each of the section cuts provides the basic data for the wing torque box weight analysis. The synthesis and weight analyses are performed for the following:

1. Upper and lower cover

2. Supporting intermediate spars or ribs

3. Front and rear spar webs

4. Load redistribution bulkhead ribs

5. Miscellaneous attachments

The cover skin thicknesses are determined for general or local instability and then adjusted if necessary for section rigidity requirements. The design restraints are included as part of the input data to achieve the required design. The skin and stringer allowable compression sheet buckling stresses are determined by

$$F_{ccr} = \frac{K\pi^2 E_r}{12\,(1-\mu^2)} \left(\frac{t}{b}\right)^2$$

Where

K = sheet buckling coefficient

E_r = reduced modules

μ = Poisson's ratio

t = sheet thickness

b = sheet support width

The general column instability stress is determined from:

$$F_{col} = \frac{\pi^2 E_t}{(L/\rho)^2}$$

Where

E_t = tangent modulus

L = effective column support length

ρ = radius of gyration of stringer column about its neutral axis

The plate and column properties in the elastic and inelastic ranges are determined from an equation fit on the stress-strain relationship of the material. For each given stress, the curve fit of strain = f(stress) permits calculation of strains, secant, and tangent modulus. These data are used in the complex search-analysis routine to equate and determine the critical stress-geometry parameters.

The weight analysis consists of calculations of individual component weights, and adjusting these weights based on a matrix of indexing coefficients for each elemental component of the synthesized structure. These coefficients are weight indexing and program calibrating coefficients which are used to correlate program estimates of representative sample wings of existing airplanes. These multiple coefficients (factors) account for the existing deficiencies in representing a real structure from an idealized structure. These factors account for items which cannot be analyzed or measured, such as local skin pads, machine runouts and thickness tapers, localized cutouts and fittings, etc. The multiple factor approach is used to minimize errors inherent in a single factor approach and also to (1) provide better correlation of individual components, (2) to insure proper variation in weights and distribution of the torque box structure for weight trade studies, and (3) to provide a means where components can be incremented for localized requirements or for unique design requirement which cannot be analyzed within the program. An example is the weight of wing cover: Each cover is indexed by four factors. The total wing cover weight equation has the form:

$$W_{cover} = K_{cover} (K_{skin} W_{skin} + K_{str/cap} W_{str/cap} + K_{misc} W_{misc})$$

9.3.2.1 Fixed Leading and Trailing Edges

The weight estimates for these components were based primarily on statistical data. The effects of STOL requirements such as blown flaps, etc, were considered by use of the multiple factors as discussed previously. The unit weights are listed in table XXXI and unique STOL feature weights are given in table XXXII.

9.3.2.2 Control Surfaces

The STOL control surfaces estimated weights are based primarily on statistical data and are incremented for differences in design, geometry, and types of supports.

9.3.3 EMPENNAGE

The horizontal and vertical tail weights were estimated in the same manner as the wing. The wing weight estimation program and procedures apply equally well to the empennage. Multiple factors were developed for the T-tail horizontal stabilizer and the double-hinged control surfaces. Indexing factors for the empennage are listed in table XXXIII.

9.3.4 FUSELAGE GROUP

The MST fuselage weight estimate was developed through the use of the weight-estimating computer program. The program incorporates analytical, empirical, and statistical methods that provide the latitude for evaluating structural components as a function of design criteria and constraints. A discussion of this estimating procedure follows.

9.3.4.1 Primary Structure

The primary shell structure estimate is based on an analytical evaluation of loads on a multistation descriptive shell model. The process also evaluates material properties and manufacturing restraints. However, to determine the inherent nonoptimum effects, and to account for the limitations of an indexing consisted of estimating the shell weight of an existing vehicle, comparing the estimates with the actual weights, and developing the correlation factors. The index factors for this configuration were derived from an evaluation of the C-141A vehicle. The C-141A was selected due to similarity with the the proposal vehicle in geometry, construction, and cargo handling systems.

The construction selected for the proposal vehicle is skin-stringer semimonocoque design. The frame spacing of 20 inches was established by the requirement for tiedown fittings on a 20-inch grid. Stringer spacing was set

TABLE XXXI. WING AND TAIL UNIT WEIGHT

Item	Area	W/S	Weight (lb)
Wing box	735	14.72	10,823
Secondary structure	-	-	1,177
Aileron	69	6.32	436
Trailing edge flaps	309	10.52	3,251
Leading edge slats	210	8.31	1,746
Spoilers	197	3.67	723
Fixed LE and TE	-	-	824
Stabilizer	365.5	5.57	2,037
Fin	256.5	7.13	1,826
Secondary structure	-	-	344
Elevator	299	2.61	780
Rudder	154	4.25	655
Tail fairing	-	-	308
Droop LE (horiz)	-	-	200

TABLE XXXII. UNIQUE WEIGHTS

Item	S	ΔW/S	Weight (lb)	Remarks
TE flap	309	4.03	1,245	Double slotted
LE flap	210	1.76	370	Powered slap
Horiz tail	665	0.19	130	Double hinge
Horiz LE	100	2.00	200	Droop LE
Vert tail	410	0.31	127	Double hinge

TABLE XXXIII. PROGRAM CALIBRATION AND NONOPTIMUM INDEXING FACTORS FOR EMPENNAGE TORQUE BOXES

Torque-Box Element	Incremental Factors	
	Vertical	Horizontal
Upper cover, average total	+0.215	+0.215
Lower cover, average total	+0.215	+0.215
Local misc shim pads, runouts, etc, average upper and lower	+0.030	+0.030
Ribs and bulkheads	+0.150	+0.150
Front spar, average total	+0.350	+0.350
Rear spar, average total	+0.350	+0.350
Misc and attachments, average total	+0.050	+0.050

at 6 inches. 7050 aluminum alloys were selected as the primary structural material for the fuselage, primarily because of better fatigue properties over the 7075 and 2024 alloys. The estimated weight savings in fuselage covers, longerons, and major frames is 150 pounds when compared to the 2024 aluminum.

9.3.4.1.1 Design Loads

The spectrum of loads investigated included 2 G taxi and 2.5 G flight maneuver at maximum gross weight, 3 G flight maneuver at primary mission design weight, and 15 feet per second STOL landing at midmission design weight. Lateral and vertical gust were also evaluated.

9.3.4.1.2 Major Frames and Bulkheads

The major load redistribution frames for wing, empennage, and landing gear support were estimated on the basis of a multicut, internal ring load evaluation. Pressure bulkhead estimates were based on conventional stiffened sheet construction. The index factor of 1.2 was applied to the theoretical frame estimates.

9.3.4.1.3 Minor Frames

The minor frames were estimated on the basis of an average frame depth of 6 inches. Minimum frame cap thickness was input at 0.050 inch and web thickness at 0.032 inch. The criteria evaluated were panel constraint and the frame stiffness required for general shell stability. The analytical estimate was multiplied by 1.226 to obtain the final estimate for minor frames. The floor frames, which are designed by local requirements, were established statistically and are discussed under paragraph 9.3.4.1.7.

9.3.4.1.4 Covering

The cover weight was determined by evaluating the panel for shear strength and cabin pressurization. The theoretical cover thicknesses that satisfied these requirements for each of four segments around the periphery of the defined section cut were assumed to be tapered between cuts. The thicknesses were then integrated and multiplied by the index factor (1.220) to obtain the estimate for the covering weight. The relatively high correlation factor for the covering reflects the inability of the programmed approach to predict doubler requirements for safe failure design of pressurized vessels.

9.3.4.1.5 Stringers

The stringer estimates were derived by establishing a maximum allowable stress level at 90 percent of compression yield strength, and a minimum stringer area of 0.145 square inch. Theoretical areas that satisfied strength and stability were multiplied by 1.182 to obtain the stringer weight. In addition to stringers, longerons were provided in the area of the wing, aft ramp, and cargo door cutouts. The stringers that end at the edges of the cutout were assumed to be of minimum area, with the longerons providing the necessary strength. The longerons extend beyond the cutout for a distance equivalent to the cutout span.

9.3.4.1.6 Joints Splices and Fasteners

Joints splices and fastener weight estimates were based on 10 percent of the cover and longeron weights. The wing and empennage attach fittings were estimated on the basis of the net fitting loads.

9.3.4.1.7 Flooring and Supports

The flooring and support structure, which are designed by local requirements such as tiedown loads, threadways, tire footprint, and 400 pounds per square inch local load, was estimated statistically. The cargo floor unit weight was based on the following equation:

$$\text{Unit weight} = 4.8 \times \left(\frac{\text{Width}}{14.16}\right)^{0.66} \text{ pounds per square feet}$$

9.3.4.1.8 Longitudinal Partitions

The partition estimates were based on stiffened sheet construction. A pressure diaphragm was provided in the wing center section cutout area. The aft deck was based on providing torsional capability to support empennage loads in the area of the ramp and aft door cutouts.

9.3.4.2 Secondary Structure

The secondary structure estimates were derived from geometrical and functional design requirements. Statistical data were used for each component, with modifications to reflect design features.

The windshield estimate was based on impact by a 4-pound bird at 388 miles per hour. The side windows were estimated to satisfy an ultimate cockpit pressure of 17.6 pounds per square inch.

Miscellaneous access door weight requirements were based on the following statistical equation:

$$\text{Misc access} = 10.45 \ (\text{Wetted Area})^{0.28}$$

The other secondary structure items were estimated on the basis of statistical unit weights. These unit weights are tabulated in table XXXIV.

Stairways and ladder requirements for the crew station were estimated to be 49 pounds. Walkways were provided on both sides of the cargo bay; the estimate for this item was 0.1 pound per linear inch of walkway. Antiskid protection was provided at 0.035 pound per linear inch.

In-flight refueling receptacle provisions were estimated to be 100 pounds. The paratroop spoiler doors were estimated to be 20 pounds.

9.3.5 LANDING GEAR

The analysis of the MST landing gears includes the consideration of the MIL 8860 series specifications and also the inclusion of the STOL sink speed requirement of 15 feet per second at midmission gross weight.

The primary loads are due to brake roll, turning, and unsymmetrical braking. The tire weights are based on tire sizing compatible with soft-field requirements of CBR 6 and 200 passes.

The weight of the basic landing gear structure was established by the use of the Mass Properties Section landing gear program. This program is capable of generating loads in accordance with MIL-A-8862, airplane strength and rigidity, and landing and ground handling loads. This program also is capable of assembling a total landing gear weight based on computed loads.

The sizing of wheels and tires was determined by using the methods shown in AFFDL-TM-71-09-FEM, "Design Procedure for Establishing Aircraft Capability to Operate on Soil Surfaces."

Table XXXV gives a summary of landing gear weights.

9.3.5.1 Ground Mobility System

The ground mobility system is an integral part of the landing gear system with the prime power source as the APU system.

The main landing gear drive train consists of a hydraulic motor, planetary gearbox, and chain drives. The hydraulic system consists of two APU-driven, 4,000 psi hydraulic pumps, reservoir, valves and controls, and the fluid distribution system.

TABLE XXXIV. SECONDARY STRUCTURE UNIT WEIGHTS

Item	Unit Weight (psf)
Flooring and support crew station	2.21
Radome	1.75
Nose landing gear door	5.0
Main landing gear door	5.3
Aft cargo door	2.9
Aft ramp	12.36
Aft ramp toe	4.0
Aft pressure door	6.2
Escape doors	15.0
Paratroop door	11.0
Entrance door	10.0
Main gear pod	2.2
Exterior finish*	0.026
Interior finish and corrosion protection	0.050

*The unit weight for exterior finish is for decals and numerals and does not include exterior paint.

TABLE XXXV. SUMMARY OF LANDING GEAR WEIGHTS

Item	D516-2A
Main	
Structure	3,545
Wheels, brake, and tires	2,754
Controls	820
Fatigue	226
Total - Main	7,345
Nose	
Structure	373
Wheels and tires	513
Controls	161
Fatigue	33
Total - Nose	1,080
Total landing gear	8,425

9.3.5.1.1 Weight Summary

Hydraulic System	Weight (lb)
Pumps (2)	60
Reservoir	10
Heat exchanger	10
Filters	40
Valves and control	10
Motor (2)	336
Plumbing	41
Fluids	57
Mechanical controls	121
Mechanical drive	185
Total	870

9.3.6 SURFACE CONTROL

9.3.6.1 Introduction

The surface control system consists of:

1. Primary flight control system

2. Secondary flight control system

3. Manual control system

4. Automatic flight control system

The baseline vehicle primary flight control system consists of linear hydraulic actuators supplied by four hydraulic systems. The system is designed for mission completion without performance degradation with one system inoperative. The stability augmentation system is a fail-operate system. Automatic pilot functions are also supplied.

The elevator and rudder control surfaces are double hinged, with the forward control panel locked out at high-speed flight.

A summary of flight control system weights is given in table XXXVI.

9.3.6.2 Primary Flight Control System

The primary flight control system consists of the horizontal stabilizer, elevators, rudders, spoilers, and aileron actuation units. The actuators are

sized to 133 percent of the maximum hinge moment to meet the one system inoperative criteria. The hydraulic system pressure is 4,000 psi with fluid transmission lines made of titanium. Coiled tubing is used for actuator hookup to the feeder line to allow for the actuator angular movement.

9.3.6.2.1 Weight Analysis

The weight of the system was derived by using the estimating procedure developed at NR and used the Flight Analysis Group design data as input data. The design data and the weight data are listed in table XXXVII.

9.3.6.3 Secondary Flight Control System

The secondary flight control system includes the flap and slat, and horizontal leading edge actuation system. These actuators are linear mechanical actuators powered by hydraulic motors through gearboxes and mechanical shafting.

9.3.6.3.1 Weight Analysis

The weight estimate includes provisions for linear ball screw actuators, gearboxes, angle drive, hydraulic power drive system, synchronizing units, shutoff control valves, electronic controllers, and mechanical shafts. The pertinent data are shown in table XXXVIII.

9.3.6.4 Manual Control System

The manual control system consists of the pilot control input devices (column, wheel, and rudder), cables, and bellcranks from the crew station to the master cylinders, and a dual push rod system from the mixer bay to the linear actuator. A tandem-balanced actuator with two hydraulic systems is used for the master cylinder for all three axes. Artificial feel and override bungees are provided.

9.3.6.4.1 Weight Analysis

The weight analysis includes provisions for a conventional control wheel and column, rudder pedals for the pilot and copilot, a 3/16-inch clad cable system, and a dual aluminum push rod system with wall thickness equal to or greater than 0.35 inch. The system weights are listed in table XXXIX.

TABLE XXXVI. SUMMARY - FLIGHT CONTROLS

Functions	Weight (lb)		
Cockpit controls			160
AFCS			335
Pitch control			643
Actuation		288	
Locks, mechanism		45	
Signal transmission		212	
Hydraulic plumbing		98	
Yaw control			307
Actuation		67	
Locks, mechanism		45	
Signal transmission		195	
Roll control			554
Aileron controls		(254)	
Actuators	38		
Signal transmission	130		
Plumbing	86		
Spoiler controls		(300)	
Actuation	77		
Electrical servo	112		
Signal transmission	111		
Flaps, trailing edge			346
Actuation		162	
Power system		183	
Slats			211
Actuation		77	
Power system		134	
Direct lift control, flaps			147
Actuation		62	
Interconnects		64	
Plumbing		21	
Miscellaneous			42
Horizontal Leading Edge			140
Actuation		56	
Power System		28	
Power Transmission		24	
Controls		10	
Electrical Provision		15	
Miscellaneous		7	
Total flight control system		2,885	

TABLE XXXVII. PRIMARY FLIGHT CONTROL SYSTEM

Functions	Control Surface				
	Stabilizer	Elevator	Rudder	Spoilers	Aileron
Hinge moment (in.-lb)	3.23×10^6	179,600	503,700	23,468	40,446
Rate, maximum (deg/sec)	10	50	50	120	100
Total deflection (deg)	30	50	50	60	50
Stroke (in.)	8	10.0	12.8	6.3-2.6	3.41
Hydraulic flow (gpm/sys/act.)	10.0	5.4	12.0	1.26-0.33	3
Required piston area (sq in.)	53.83	1.968	2.763	0.38-0.25	[illegible].25
Actuation weights (each) (lb)					
Actuator	83.1	3.4	8.8	1.5	1.9
Valve	9.0	4.0	3.8	9.4	2.3
Fluid	7.9	0.6	1.4	0.2	0.1
Flex connection	8.0	0.6	0.9	0.3	0.2
Miscellaneous	16.2	1.3	1.8	1.4	0.7
Total	124.2	9.9	16.7	12.8(4)	5.2
Number of actuator/ship	2.0	4.0	4.0	6.0	6.0
Hydraulic system (No.)	4.0	4.0	4.0	3.0	3.0
Vol/actuator (cu in.)	33.1	2.74	11.9	1.2	1.5
Primary flight control Actuation system weight (lb)	248.0	39.6	66.8	77	31.2

*For four panels (four actuators)

TABLE XXXVIII. SECONDARY FLIGHT CONTROL SYSTEM

Functions	Flaps	Leading Edge Slats
Area/side (square feet)		
No. 1	79.3	18.9
No. 2	44.9	23.0
No. 3	35.6	27.0
No. 4	37.1	35.0
No. 5	--	
No. 6	--	
Weight data (pounds)		
Actuators	163	77
Hydraulic motor	15	15
Gearbox	40	30
Control(s), hydraulic	10	10
Transducers	5	5
Controller	10	--
Mechanical shafting	90	64
Misc	13	10
Total system weights	346	211

TABLE XXXIX. MANUAL CONTROL SYSTEM

Items	Weight (lb)
Cockpit controls	160
Signal transmission	
Pitch	212
Yaw	195
Roll	241
Spoiler, electrical signal provision	113

9.3.6.5 Automatic Flight Control System (AFCS)

The AFCS provides the autopilot mode and stability augmentation system. The augmentation system is a fail-operate system, including redundant monitors and controllers, dual servoactuators, and dual control signal transmission system. Motion sensors (accelerometers, gyros), a control air data computer, and a flight director computer are provided.

9.3.6.5.1 Weight Analysis

The weight of the system was based primarily on the B-1 system concept, except removal of the SOFTRIDE features. Component weights are generally B-1 weights. The wiring and installation weight includes provisions for redundancy. The system weights are listed in table XL.

TABLE XL. AFCS SUMMARY

Item	Weight (lb)
Controllers, autopilot	40
SAS controllers	
Pitch	30
Roll	30
Yaw	30
Gyro platform	25
Control panel	10
Sensors	15
Transducers	30
Angle-of-attack transmitter	5
Computer control panel	2
Misc controls	6
Wiring and installation	77
Cooling provision	10
Support	25
Total AFCS	335

9.3.6.6 STOL Penalties

An estimate of STOL requirement which would add to a CTOL airplane is listed as follows:

	Weight (lb)
Horizontal stabilizer	
Stabilizer actuator	109
Linkages and locks	39
Leading edge actuation	140
Rudder System	
Actuators	35
Linkages and locks	35
Hydraulic power supply	90
Total	448

9.3.7 NACELLE GROUP

Table XLI presents the nacelle group dimensional data and weight breakdown for the MST configuration.

The nacelle and pylon weight estimates were based on the specific weights (pounds per square foot) shown in table XLI, adjusted from C-141 empirical data, taking into account the difference in overhang and hung weight on the pylon weight.

9.3.8 PROPULSION GROUP

Table XLII presents the D516-2A propulsion data, while table XLIII presents the weight statement.

9.3.8.1 Engine Installation

9.3.8.1.1 Engines

The 93 percent GE13/F-10 engines, with a thrust of 22,320 pounds, weigh 2,740 pounds each at a T/W = 8.1:1.

9.3.8.1.2 Thrust Deflectors

The thrust deflectors (swivel nozzle) weight is based on an equivalent thickness (t) of a 0.080-inch titanium sheet of frustrum shape with a base diameter of 72 inches, an exit diameter of 55 inches, and a length of 48.8 inches. It has a total area of [illegible] square feet, which results in a specific weight of 1.84 pounds per square foot. In addition, a weight of 73 pounds per engine is included for hinge fittings, bearings, actuators, actuation mechanism, and plumbing.

9.3.8.1.3 Thrust Reversers

The thrust reverser weight is taken as 15 percent of the engine weight based on statistical data. They are of the cascade type, and the reverse thrust exit doors are on only the upper quadrant of the nacelle rather than around the complete circumference.

TABLE XLI. NACELLE GROUP DATA

Item	
• Nacelle Dimensional Data	
Number required	4
Average diameter (inches)	78
Length (inches)	144
Area (square feet)	251.33
Specific Weight (pounds per square foot)	2.605
• Pylon Dimensional Data	
Number required	4
Inboard pylon projected area (square feet))	38
Inboard pylon specific weight (pounds per square foot) (projected)	11.774
Outboard pylon projected area (square feet)	38
Outboard pylon specific weight (pounds per square foot) (projected)	11.774
• Nacelle Group Weight Breakdown (pounds)	(4,410)
Nacelles (4)	2,620
Inboard pylons (2)	895
Outboard pylons (2)	895

TABLE XLII. PROPULSION GROUP DATA

Item	
Number of engines	4
Engine designation	GE13/F-10
Engine Scaling (percent)	93
Engine thrust (SLS) (pounds)	22,320
Engine thrust/weight ratio	8.1:1
Fuel (JP-4) max flow rate (pounds/hr/engine (ω))	8,930
Fuel weight (max) (pounds)	63,992
Fuel weight - design mission (pounds)	33,161
Wing area - total (square feet)	1600
Engine exhaust system	
Tail cone, thrust deflectors, thrust reversers on fan only	
Fuel tank pressure (ΔP = 1.5 psi) (psi)	16.2
Fuel tank purging and inerting	LN_2
Starting system type	
Bleed air from main engines, APU bleed air, or ground supply to Airesearch type ATS-100 air turbine on engines	

TABLE XLIII. PROPULSION GROUP WEIGHT STATEMENT

Item	Weight (lb)
Propulsion Group	(17,350)
Engine installation	(14,445)
Engine - bare	10,960
Thrust deflectors	883
Thrust reversers	1,642
Tail cone	445
Engine shroud	315
Residual fluids	200
Fuel system	(2,365)
Main feed subsystem	246
Fuel transfer subsystem	217
Refueling subsystem	205
Vent subsystem	203
Dump subsystem	107
Pressurization and inerting subsystem	158
Fuel fill level control subsystem	136
Drain subsystem	19
Fuel tank sealant	858
Fuel tank boards, supports, and bulkheads	216
Engine controls	(125)
Consoles (4)	80
Linkage	35
Misc	10
Starting system	(415)
Air turbines (4)	105
Control valves (5)	50
Ducting	180
Bellows and connecters	40
Supports	10
Electrical provisions	30

9.3.8.1.4 Tail Cone

The tail cone spike assumes an equivalent thickness (t) of 0.060-inch titanium and has a surface area of 23.94 square feet. This results in a specific weight of 1.38 pounds per square foot. Since the cone is adjustable fore and aft, a weight allowance was provided for tracks and an actuating system.

9.3.8.1.5 Engine Shroud

The engine shroud weight assumes an equivalent thickness (t) of 0.040-inch titanium and has an area of 72.95 square feet. This results in a specific weight of 1.10 pounds per square foot, including attachment provisions.

9.3.8.1.6 Residual Fluids

The residual fluid weight is estimated at 50 pounds per engine.

9.3.8.2 Fuel System (Refer to Table XLIV for Weight Breakdown)

The fuel system has been broken down into eight subsystems, plus fuel tank sealants. This system weight was estimated from a schematic (refer to paragraph 10.5) using scaled B-1 fuel system component weight information.

All plumbing and most line installed items for the 93 percent GE13/F-10 engines were scaled from previous sizing that had been made for a 107.5-percent GE13/F6A engine which had a maximum fuel flow rate of 8,600 pounds per hours per engine. The 93 percent GE13/F-10 engine has a maximum fuel flow rate of 8,930 pounds per hour per engine. Thus assuming the same flow velocities, the plumbing cross-section area would be increased by the ratio of

$$\frac{8,930}{8,600} = 1.04 \text{ or 4-percent increase in area}$$

This results in an average increase of 2 percent in plumbing diameter.

9.3.8.2.1 Main Engine Feed Subsystem

This subsystem consits of four main pumps, one each in four independent sump tanks in the wing center section. The pump weight is solved by equating (by computer program CURVFT) the empirical curve of the B-1 pumps by the

TABLE XLIV. FUEL SYSTEM WEIGHT BREAKDOWN

Item	Weight (lb)
Fuel System Total	(2,857)
Main engine feed subsystem	(246)
Pumps (4)	27
Check valves (20) (incl 12 in. ribs at 0.4 lb)	22
Shutoff valves (8) (elect. sol)	25
Pressure switch (4)	2
Filter (4)	45
Pressure relief (2)	10
Plumbing (incl ftg and sup)	81
Supports (equip. less plumbing)	16
Electrical wire harness	10
Electrical equip.	8
Transfer subsystem	(217)
Pumps (6)	40
Check valves (8)	11
Flow sensors (2)	17
Plumbing (incl ftg and sup)	81
Supports (equip. - less plumbing)	16
Electrical wire harness	24
Ejector pump with check valve (4)	4
Electrical equip.	20
Cargo fuel receptacle (bayonet-type)	4
Refueling subsystem	(205)
Shutoff valve (2) (elect. sol)	6
Check valves (2)	4
Pressure switch (1) (incl in recp)	-
Gravity refueling caps and adapters (12)	24
Receptacle - servicing (2)	16
Receptacle - aerial refueling	65
Aerial refueling mechanism	29
Plumbing (incl ftg and sup)	44
Supports (equip. - less plumbing)	8
Electric wire harness	4
Electrical equip.	5

TABLE XLIV. FUEL SYSTEM WEIGHT BREAKDOWN (CONT)

Item	Weight (lb)
Vent subsystem	(203)
Vent float valve (16)	7
Check valves (3)	3
Pressure relief valve (2)	11
Vent valve and controls	35
Plumbing (incl ftg and sup)	115
Supports (equip. - less plumbing)	6
Electrical wire harness	16
Electrical equip.	10
Dump subsystem	(107)
Shutoff valve (2)	7
Flow sensor (2)	18
Plumbing (incl ftg and sup)	67
Supports (equip. - less plumbing)	6
Electric wire harness	4
Electrical equip.	5
Pressurization and inerting subsystem	(158)
Tank - LN_2 (184 lb LN_2 capacity)	79
Frangible disc. (1)	1
Pressure relief (1)	5
Check valve (1)	1
Pressure gage (incl in press. cont)	-
Inlet panel or perforated line	5
Plumbing (incl ftg and sup)	20
Supports (equip. - less plumbing)	16
Electrical wire harness	8
Electrical equip.	10
Misc	13
Fuel fill level control subsystem	(136)
Level cont valve, duel remote actuating (8)	40
Level cont valve, pilot-dual with precheck (8)	40
Electric wire harness	31
Plumbing (incl ftg and sup)	10
Electrical equip.	8
Misc	7

TABLE XLIV. FUEL SYSTEM WEIGHT BREAKDOWN (CONCL)

Item	Weight (lb)
Drain subsystem	(19)
Shutoff valve (1) (elect. sol)	1
Plumb, scavenge (1)	6
Flow sensor (1)	1
Plumbing (incl ftg and sup)	3
Supports (equip. - less plumbing)	1
Electrical equip.	5
Misc	2
Fuel tank sealant	(858)
Normal sealant (faying and fillets) (0.02 lb/gal.)	197
Self-sealant (1.8 lb/ft^2) (bottom and sides of tanks)	477
Bladder cells	184
Fuel tank backing board, support, and bulkheads	216

orthogonal polynomial technique and takes the form of a 5th degree regression formula,

$$W_p = -1.09 \times 10^{-5} (\omega/4000)^5 + 4.85 \times 10^{-4} (\omega/4000)^4 - 7.07 \times 10^{-3} (\omega/4000)^3$$

$$+ 5.046 \times 10^{-2} (\omega/4000)^2 + 5.67 \times 10^{-2} (\omega/4000) + 6.314$$

where:

W_p = pump weight (pounds)

ω = fuel flow rate (pounds per hour)

At the flow rate of 8,930 pounds per hours, this results in a pump weight of 6.7 pounds.

The plumbing for the main engine feed consists of 160 feet of 2-1/4 x 0.035 inch aluminum lines plus fittings every 5 feet and rib fittings based on 22-inch rib spacing. Also included are associated valves fittings, supports, and electrical provisions.

9.3.8.2.2 Transfer Subsystem

This system consists of six transfer pumps, three in each wing; and four ejector pumps, two in each wing. The main transfer pumps are identical to the pumps in the main engine feed subsystem.

The plumbing consists of the following:

47 feet of 3-1/2-inch-diameter x 0.042-inch aluminum tubing

45 feet of 2-1/4-inch-diameter x 0.035-inch aluminum tubing

83.4 feet of 9/16-inch-diameter x 0.028-inch aluminum tubing

Allowance for fittings every 5 feet plus rib fitting supports based on 22-inch rib spacing. Also included are valves, sensors, supports, electrical provisions, and a receptable for connection to cargo fuel.

9.3.8.2.3 Refueling Subsystem

This subsystem consists principally of valves, servicing and aerial refueling receptacle, and actuation mechanism. Plumbing consists of 39.6 feet of 4-inch-diameter x 0.049-inch aluminum tubing, plus fittings and rib support fittings. Also included are 12 gravity refueling caps and adapters, supports, and electrical provisions.

9.3.8.2.4 Drain Subsystem

This subsystem consists of one scavenge pumps, shutoff valve, flow sensor, plumbing, supports, and electrical provisions. The plumbing consists of 5.8 feet of 2-inch-diameter x 0.035-inch aluminum tubing, plus fittings and rib support fittings.

9.3.8.2.5 Vent Subsystem

The vent subsystem consists of valves, supports, plumbing, and electrical provisions. The plumbing consists of the following:

46.6 feet of 4-1/2-inch-diameter x 0.049-inch aluminum tubing

41.5 feet of 3-1/2-inch-diameter x 0.042-inch aluminum tubing

122.5 feet of 1-1/2-inch-diameter x 0.035-inch aluminum tubing

9.3.8.2.6 Dump Subsystem

This subsystem ties into the transfer system and consists of plumbing, valves, sensors, supports, and electrical provisions. The plumbing consists of 85 feet of 3-1/2-inch-diameter x 0.042-inch aluminum tubing, plus fittings and rib support fittings.

9.3.8.2.7 Pressurization and Inerting Subsystem

The LN_2 requirements of 184 pounds (3.7 cubic feet) is based on the following criteria:

1. Inerting volume of 1,316.2 cubic feet based on maximum JP-4 fuel weight 64,000 pounds at a density of 48.6 pounds per cubic feet

2. Fuel tank ΔP of 1.5 psi or 16.2 psi for N_2 determination

3. Temperature minimum at empty condition of -65° F

4. 48-hour boil-off plus lines and ullage = 20 pounds

5. LN_2 stored subcritically at 100 psi operating, 200 psi proof, and 300 psi burst (density of LN_2 50.26 pounds per cubic feet)

The LN_2 tank weight was scaled from the B-1 tank having a double wall aluminum construction. A 25-percent increase in weight over direct proportion to capacity of the B-1 tank was applied due to the smaller capacity of the STOL tank. Also included in the LN_2 subsystem were valves, plumbing, frangible disc, inlet panel or perforated lines, supports, and electrical provisions. The vent lines might be used for the inerting and pressurization. The additional plumbing consists of:

16.7 feet of 1-1/2-inch-diameter x 0.035 aluminum tubing

51.2 feet of 1-inch-diameter x 0.028 aluminum tubing

plus fittings and rib support fittings.

9.3.8.2.8 Fuel Fill Level Control

This subsystem consists of eight dual remote actuating level control valves and eight dual pilot level control valves, plus plumbing and electrical provisions. The plumbing consists of 140 feet of 3/8-inch-diameter x 0.028 aluminum tubing plus fittings and supports.

9.3.8.2.9 Fuel Tank Sealant

The fuel tanks sealant consists of the normal faying and fillet sealant at 0.02 pound per gallon of fuel, and the self-sealant at 1.8 pounds per cubic foot. The latter is assumed for the bottom and sides of the wing tanks from station 0 to station 263 (approximately). This sealant is a sandwich of 3/8-inch foam with 1/16-inch, two ply nylon face sheets plus 0.1-inch bonding, giving a total thickness of 0.6 inch. Total area is approximately 319 square feet, weighs 477 pounds, and displaces approximately 756 pounds of fuel.

9.3.8.3 Engine Controls

This weight estimate was taken from the NR 260 STOL medium tactical transport study and consists of four throttle consoles, linkage to engines, and misc.

9.3.8.4 Starting System

This system consists of four engine-mounted AiResearch-type ATS-100 air turbine starters (at 26.5 pounds each) and control valves (four at 10 pounds each) connected to the APU by 2,000 inches of 4-inch-diameter x 0.025-inch steel duct (180.0 pounds) plus a 50-pound provision for connecters, bellows, and supports. It is estimated that this duct will be exposed to 110 psi and 600° F air. There is also included a ground air supply connection and provisions.

9.3.8.5 STOL Penalty

This estimate for STOL penalty is for the thrust deflector nozzle system. These defectors are used during takeoff and landing approach, and have the capability of deflecting the exhaust stream ±15 degrees from the cruise angle.

The approximate penalty is 320 pounds per airplane.

SECTION X

SUBSYSTEM DEFINITION

The subsystem analysis contained in this section was performed using the preliminary baseline configuration as defined during the part I study phase. Each of the major subsystems has been defined, as a minimum, to the extent necessary to support a realistic weight evaluation of the configuration and to permit realistic scaling, where necessary, to support the required trade studies. The flight control system discussed in this section applies to the part I preliminary baseline configuration, with the final flight control system presented separately in volume V.

10.1 AVIONIC SYSTEM

The equipment listed in table XLV was selected with consideration given to the desired avionics functions (including optional considerations) that would be in production during the 1974-1975 time period. Selection was based on experience gained during previous NR aircraft programs, including STOL transport studies. Included in the list is equipment for which provisions are recommended; that is, equipment which the aircraft should be designed to accept, even though it may not necessarily be installed on all aircraft.

With the possible exception of the navigation displays and unique data processing equipment, all of the equipment listed in table XLV are either in production at the present time or will be by the 1974-75 time period, thereby reducing development costs to a minimum. When the data processing equipment becomes better defined, it is anticipated that the majority of it will also be "off-the-shelf" and very little new design will be required.

The equipment listed in table XLV represents a good baseline to configure the STOL TAI vehicle at this time; however, it is recommended that some avionics areas require further study in future programs. These include the use of a satellite communication system and the possible deletion of the HF transceiver, implementation of an automatic TACAN updating of the navigation system, further definition of landing or airdrop requirements (possible incorporation of a CLASS beacon system), possible incorporation of Omega, the need for the stationkeeping system, and optimizing the data processing system. The results of these studies could result in a more cost-effective avionics system.

Advantage is taken of current production programs to provide a small self-contained Doppler-inertial navigation system accurate to approximately 1-nautical-mile CEP per hour. This recommended system provides global self-contained navigation and a variety of backup navigation modes; its airborne alinement capability provides the convenience of fast takeoffs without the need for ground power and lengthy alinement procedures.

TABLE XLV. REPRESENTATIVE STOL TAI AVIONICS

Function	Identification	Vol (ft^3)	Weight (lb)	Pwr AC (va)	Pwr DC (watts)
Communications					
UHF/AM radio	AN/ARC-156	0.6	35	700/100	21
UHF/AM radio (backup)	AN/ARC-150	0.2	8		60/30
VHF/AM radio	AN/ARC-115	0.2	10		25
VHF/AM radio	AN/ARC-114	0.2	7		25
Voice scrambler (2)	KY-28	0.7	38		66
Interphone (3 stations)	AN/AIC-25	0.2	21		75
PA systems (2 speakers)	AN/AIC-13	0.2	25		10
Crash beacon/recorder	AN/ORT-26 (mod)	1.55	30	60	
Radar beacon	SST-133X (mod)	0.03	4.5		35
Identification					
A/G transponder	AN/APX-101	0.3	14		30
Mode 4 computer	KIT-1A/TSEC	0.3	14		30
Navigation					
VOR/localizer (with glide slope and marker beacon)	AN/ARN-80	0.1	6		55
ADF	ARC-40-C (commercial)	0.4	19		40
UHF/DF	AN/ARA-50	0.5	17		85
TACAN	AN/ARN-91 (V)	0.7	37	340	
LORAN C/D	AN/ARN-101 (less computer)	0.6	53	158	
TALAR	AN/APN-97	0.2	6		17
Doppler radar	SKD-102 (CSA)	1.1	36	130	
Inertial nav sys	AN/ASN-109	0.8	46	300	
Adverse weather aerial delivery system avionics					
Stationkeeping	AN/APN-169	1.6	57	230	112
Radar	AN/APQ-126 (mod)	11.3	250	2,500	250
Radar altimeter	AN/APN-194	0.1	7	45	
Data Processing					
Computer	AP(1)-IBM	0.9	49	250	
Signal converter		0.5	20	130	
Integrated data entry panel		0.2	10	45	
Computer display panel		0.3	3	80	
CITS control and status panel		0.3	10	150	
CITS tape recorder		0.4	15	125	
CITS signal conditioners		0.5	20	40	
Miscellaneous					
Interface electronics		0.5	32.5	117	39
Total		25.51	900.0	4,600	945
Provisions Only					
HF/SSB radio	AN/ARC-123	2.6	80	1,390	40
Voice scrambler	KY-65	0.2	4		

The functions required of the radar include safe letdown and landing approach to a forward base, identification of the landing or drop area, navigation position fixing and fixing on identification points (IP's), beacon tracking, terrain following, and weather mapping. These requirements can best be met by a simple, reliable, conventional K_u-band radar.

The terrain-following task is needed for letdown and for low-altitude flight in adverse weather. However, it does not appear that a separate terrain-following radar is required, based on the STOL TAI performance requirements. Instead, this function can be implemented in the mapping radar with adequate performance and reliability.

A representative radar that could be used for STOL TAI is a hybrid version of the APQ-126 and APQ-139, which have components that are mainly interchangeable. The larger antenna (29 inches wide) of the APQ-139 is used to provide a 1.8-degree beamwidth, while the moving target indicator processor of this set is deleted as being unnecessary. All the desired functions are provided by this radar at a weight of approximately 250 pounds and 2,500 volt-amperes of required electrical power. The system uses conventional, noncoherent processing and operates at K_u-band.

If further operational analysis indicates that better resolution is required for landing or for precision airdrops, then the additional weight, volume, and cost of the APQ-122(V)1 may be justifiable. However, other new developments in weapon delivery should also be considered, such as the close air support system (CLASS), which was developed by Litton Industries under Air Force contract and is now being tested at Holloman AFB by the Air Force. This system uses ground-based DME under the control of a forward air controller (FAC) to update the inertial system via a secure data link, and it is providing extremely accurate weapon delivery. This system would have a direct application for the STOL TAI precision airdrops and blind landings.

Provisions are recommended for incorporating either a CLASS beacon system, a tactical loran receiver, or an Omega receiver (but not all three). A Loran set could be incorporated for navigation where Loran signals are available. The Loran computation would be performed in the computer of the data processing system. An Omega set could replace the Loran set if future events prove that accuracy of Omega is sufficient for the STOL TAI, as Omega has the advantage of worldwide coverage. The system choice will depend upon which concept the services select.

10.2 CREW STATION

The crew station includes seats, control and display panels, and work areas for the pilot, copilot, flight engineer, and navigator. Crew vision,

entry provisions, and other crew accommodations are described herein and are shown in figures 114 and 115.

Noises and extreme temperatures from sources outside the crew cabin are attenuated by insulation blankets. Temperature control of the interior is provided by crew selection of the desired temperature for cabin pressurization air. An open-loop oxygen-generating system (OLOGS) provides the standby oxygen capability with minimized logistics and maintenance requirements. The OLOGS is a self-contained oxygen system generating its oxygen supply from engine bleed air.

Emergency exits and descent devices are provided on both sides of the crew cabin, for descent to the ground. All materials used for crew cabin interiors are nonflammable or self-extinguishing and conform to the current industry standards. Portable, hand-held fire extinguishers are provided. Crew cabin floors in all traffic areas have a durable, nonskid, nonabsorbent covering. Storage for survival equipment and first aid kits is provided in readily accessible locations.

External vision for the flight crewmembers exceeds the requirements for bomber/transport aircraft as defined in MIL-STD-850D. Figure shows the vision plot from the pilot's station. Similar vision is provided from the copilot's station. Since 22 degrees of over-the-nose vision is provided, improving to 34 degrees down on the side, there is ample vision to maintain the landing touchdown spot in view at all times during the landing approach. Upward vision of more than 60 degrees is provided from 5 degrees right azimuth and 105 degrees left azimuth, measured from the pilot's eye position, and more than 70 degrees up-vision is provided from 20 degrees left azimuth to 90 degrees left azimuth. This means that the aircraft could be rolled to a 60-degree bank angle, and the horizon would still be visible from straight ahead of the airplane nose to 105 degrees left azimuth without the pilot moving his eye position.

Lighting for the crew cabin and for control panels and emergency lighting is in accordance with specification MIL-L-6503G and any specific requirements of the airplane. A minimum number of lights is connected to the APU electrical distribution system.

Crew sustenance provisions adequate for all missions are an integral part of the crew cabin. A build-in, gravity-flow potable water system provides liquids required for drinking water, instant coffee, and powdered fruit juice drinks. A hot cup provides hot water for coffee. Adequate storage areas are provided for snack meals or box lunches carried on operational and ferry missions, and for paper cups, towels, and other required accessories. Food preparation space and containers for paper trash and for liquid disposal are a part of the galley. The potable water system is capable of being serviced from the aircraft exterior by standard service carts.

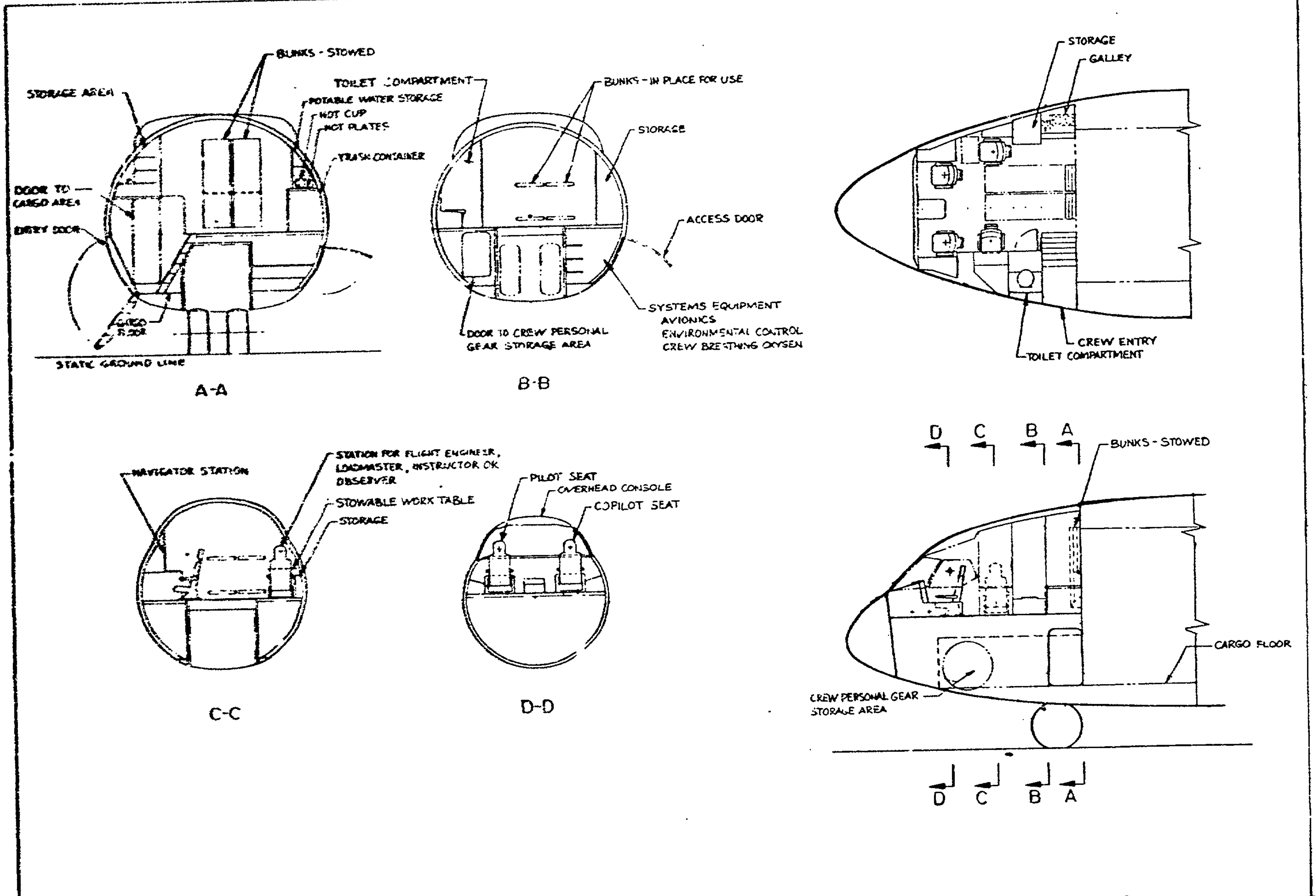

Figure 114. Crew Accommodations

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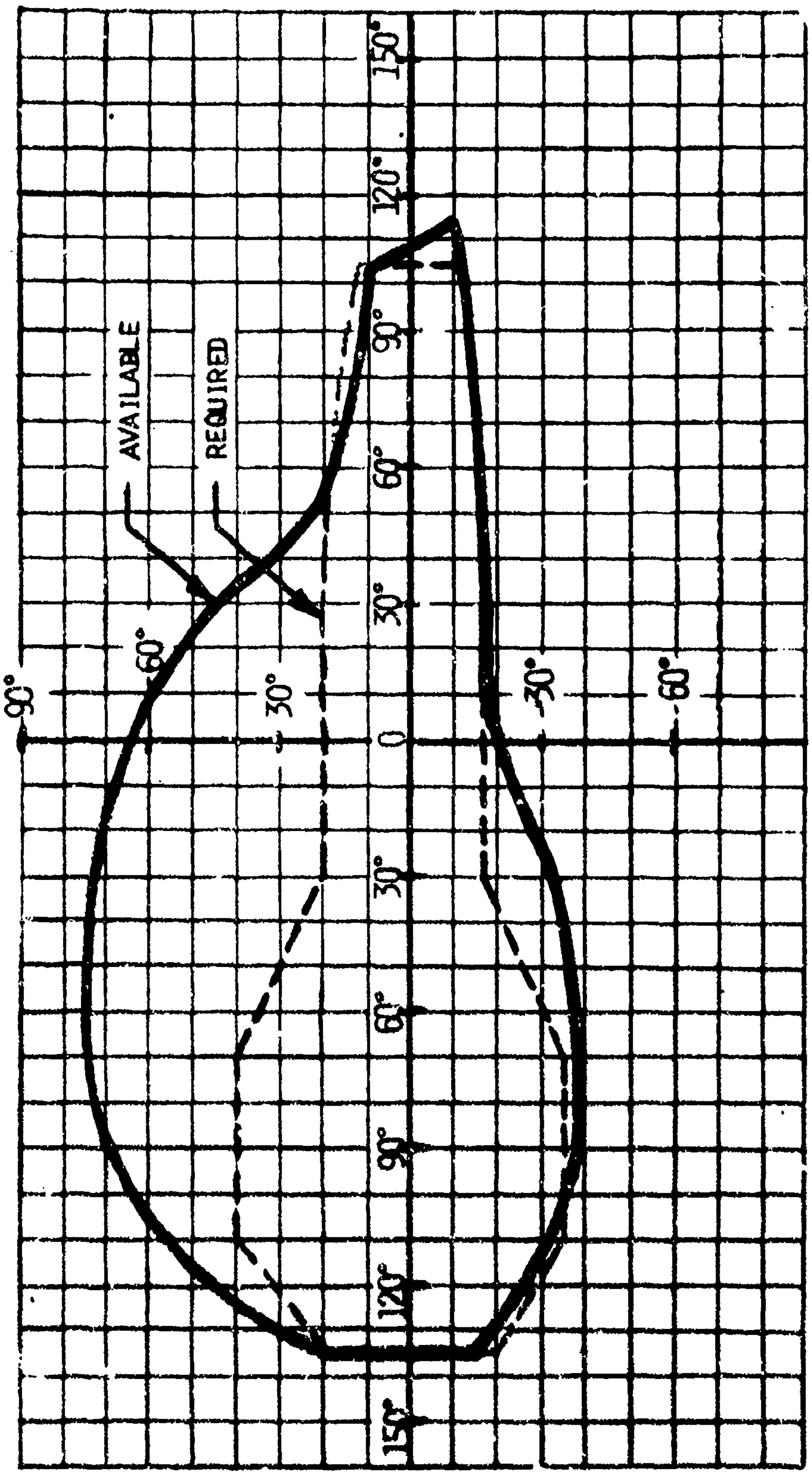


Figure 115. Pilot Vision

A recirculating toilet with a single unit containing the holding tank, bowl, and pump is provided. Servicing is from outside the aircraft by a standard toilet servicing cart.

Storage areas are provided for inclement weather clothing and other personal equipment for the crewmembers.

10.3 ENVIRONMENTAL CONTROL SYSTEM

The schematic diagram, included as figure 116, shows the selected environmental control system for the STOL TAI airplanes. Engine bleed air is taken from the intermediate stage of each engine (high stage when intermediate stage pressure is inadequate) and routed through a precooler which reduces the temperature to acceptable levels for the air-conditioning systems and aircraft protection systems. Engine fan air is used as the cooling air for the precoolers. The air is then routed through the air-conditioning packages, which consist of two identical large packs of approximately 200 pounds per minute capacity and a small pack of approximately 26 pounds per minute capacity. The packs are all bootstrap units where the cooling turbine drives both a compressor and a fan, which assists in drawing ram air and also in providing cooling air for the intercooler during ground operation. This type of unit is presently in service on several new large commercial aircraft and several small general aviation jet-type aircraft. The conditioned air passes through a water separator to reduce the free water. The cold dry air is mixed with hot precooler air to the temperature required by the cabin or cockpit or avionics temperature controller and is delivered to the proper airflow distribution system.

Table XLVI lists the major components of the basepoint environmental control system and shows the number of units, weight per unit, and total subsystem weight. Table XLVII indicates the system performance.

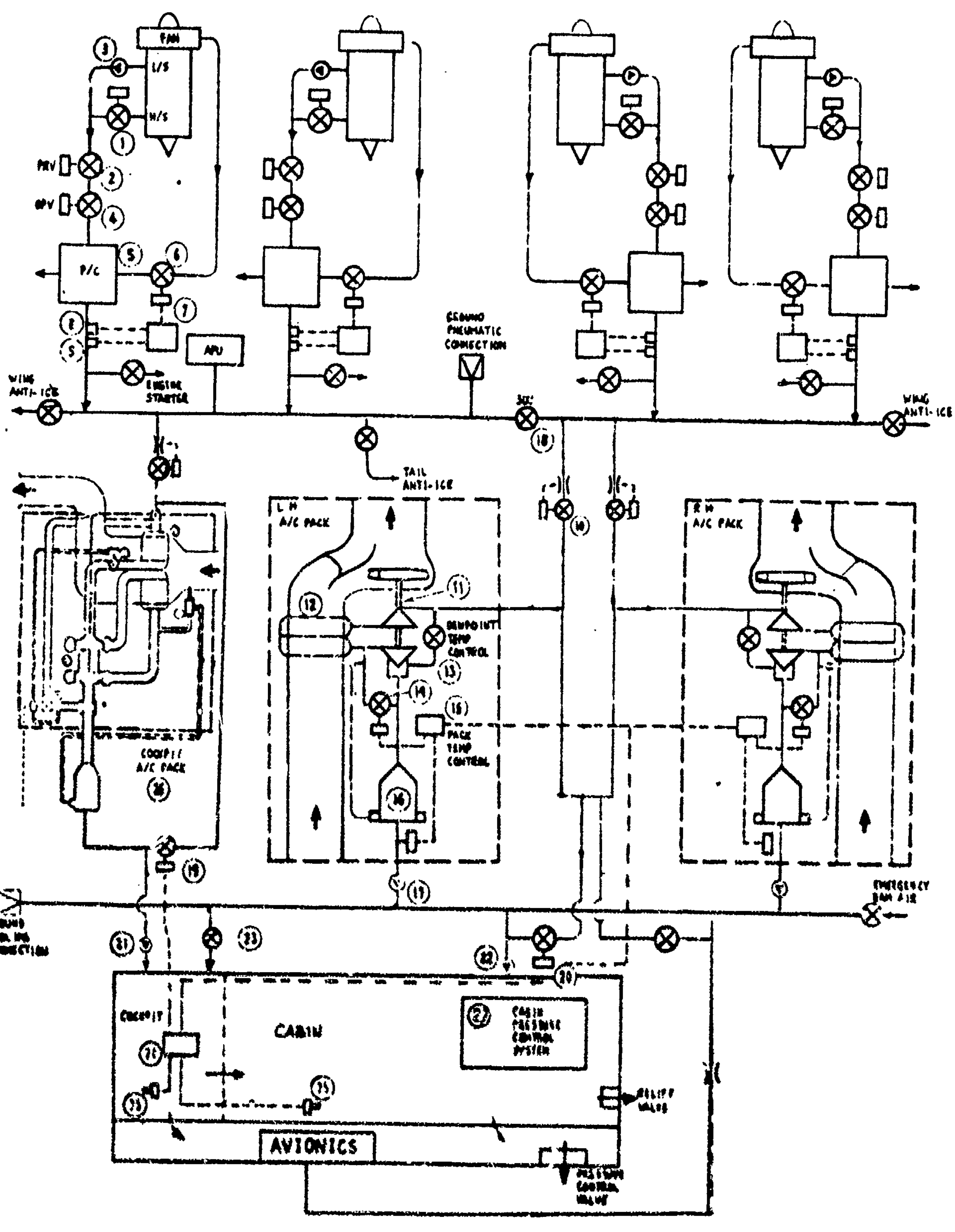


Figure 116. Environmental Control System Schematic Diagram

TABLE XLVI. ENVIRONMENTAL CONTROL SYSTEM ESTIMATED WEIGHT

Item Number	Description	No. Per Subsys	Unit Weight (lb)	Subsys Weight (lb)
	Bleed-Air Control			
1	HP bleed SO & pressure regulator valve	4	6.4	25.6
2	HP/LP bleed SO & pressure regulator valve	4	6.4	25.6
3	LP check valve	4	2.5	10.0
4	Overpressure SO valve	4	6.1	24.4
5	Precooler	4	23.5	94.0
6	Fan-air SO & temperature control valve	4	6.7	26.8
7	Temperature controller	4	2.9	11.6
8	Temperature sensor	4	0.11	0.44
9	Temperature anticipator & high-limit sensor	4	0.34	1.36
	Air-Conditioning System			
10	Flow control & SO valve, cabin	2	9.55	19.1
11	Air cycle machine, cabin	2	27.6	55.2
12	Pack heat exchanger, cabin	2	39.1	78.2
13	Water separator anti-ice valve, cabin	2	2.1	4.2
14	Pack temperature control valve, cabin	2	3.6	7.2
15	Pack temperature control, cabin	2	3.4	6.8
16	Water separator, cabin	2	11.5	23.0

TABLE XLVI. ENVIRONMENTAL CONTROL SYSTEM ESTIMATED WEIGHT (CONT')

Item Number	Description	No. Per Subsys	Unit Weight (lb)	Subsys Weight (lb)
17	Pack outlet check valve, cabin	2	0.62	1.24
18	Wing isolation valve	1	5.4	5.4
19	Valve, cockpit temperature control	1	3.25	3.25
20	Valve, cabin temperature control	2	3.6	7.2
21	Cockpit check valve	1	0.50	0.50
22	Cabin check valve	2	0.62	1.24
23	Cockpit alternate air valve	1	3.25	3.25
24	Temperature control, cabin & cockpit	1	7.5	7.5
25	Temperature sensor, cabin & cockpit	2	0.10	0.20
26	Cockpit Refrigeration Package			
	Flow control, SO & reg valve, cockpit	1	5.3	5.3
	Air cycle machine, cockpit	1	12.3	12.3
	Heat exchanger assembly	1	12.1	12.1
	Water separator, anti-ice valve, cockpit	1	3.95	3.95
	Water separator, anti-ice control, cockpit	1	0.50	0.50
	Sensor	1	0.18	0.18
	Water separator	1	2.25	2.25
	Compressor inlet throttle valve	1	5.3	5.3
	Aspirator	1	0.1	0.1
	Plenums, ducts, couplings	1	--	4.0

TABLE XLVI. ENVIRONMENTAL CONTROL SYSTEM ESTIMATED WEIGHT (CONCL)

Item Number	Description	No. Per Subsys	Unit Weight (lb)	Subsys Weight (lb)
27	Cabin Pressure Control System			
	Actuator, rotary cabin outflow valve	1	6.28	6.28
	Valve, safety, cabin pressure	2	6.2	12.4
	Sensor, differential pressure	1	1.0	1.0
	Amplifier, electronic control	1	2.1	2.1
	Air filter	1	0.3	0.3
	Controller, cabin pressure	1	1.95	1.95
	Relay, time delay	1	0.4	0.4
28	Flow control	1	4.0	4.0
29	Temperature control	1	1.0	1.0
	Total weight			518.69

10.3.1 ADVERSE WEATHER PROTECTION

For the purpose of this study, wing, tail, and cowl anti-icing were considered as being a hot-air system using 320 pounds per minute total, or 1.25 pounds per second per engine of bleed airflow. It was assumed that airframe anti-icing would not be used when bleed air was being used for lift purposes. The anti-icing were considered in sizing of the precoolers (item 5 of table XLVI).

Windshield anti-icing and defogging are provided by an electric conductive film which is an integral part of the panels being protected.

Rain removal consists of conventional windshield wipers.

TABLE XLVII. ENVIRONMENTAL CONTROL SYSTEM PERFORMANCE

Conditions						
Operation phase	Ground	Idle	Takeoff	Landing	Cruise	Loiter
Altitude	SL	SL	SL	SL	35,000	SL
Mach No.	0	0	0.20	0.15	0.8	0.3
Day	Hot	Standard	Hot	Hot	Standard	Standard
Amb press. (psia)	14.7	14.7	14.7	14.7	3.46	14.7
Amb temp (° F)	103	59	103	103	-65.8	59
Amb humidity (gr/lb)	130	75	130	130	0	75
Bleed-air stage	(APU)	H/S	1/S	1/S	H/S	H/S
Bleed-air press (psia)	41	41	118	82.5	100	100
Bleed-air temp (° F)	430	300	665	540	700	540
A/C pack inlet temp	430	300	450	450	450	450
Estimated performance						
Cockpit unit						
Unit airflow (lb/min)	24.9	27.6	25.5	25.5	* 20	30.7
Supply air temp (° F DAR)	33.6	25.5	36	36	35	25.3
Supply air temp (° F DB)	49.8	33.6	51	51	35	33.3
Compt. temp (° F)	70	70	70	70	70	70
Cabin units (2)						
Supply airflow (lb/min)		147.5	170	172.5	102	173
Supply air temp (° F DAR)		46	41.5	42	39.5	30.5
Supply air temp (° F DB)		46	46	47	39.5	34.1
Compt temp, ° F		70	70	70	70	70
Total airflow (lb/min)	24.9	175.1	195.5	198	142	203.7
Ventilation rate (cfm)	332	2,330	2,600	2,640		
Engine fan flow lb/min per engine	0	0	23.5	13	22	12

*Inlet pressure regulated to 16 psig to prevent overspeed.

10.3.2 CABIN PRESSURIZATION

The cabin will be capable of being maintained at a sea-level pressure up to 23,000-foot airplane altitude and an 8.8 psi differential above that altitude up to the airplane ceiling. In addition, the cabin pressure will be selectable between -1,000 to 8,000 feet within the limits previously defined.

10.3.3 BASELINE ENVIRONMENTAL CONTROL SYSTEM ADVANTAGES

The baseline ECS provides several advantages. Utilizing a small turbomachine for the crew compartment permits use of an APU size which will not be dictated by ECS requirements. The use of two identical units for conditioning the cargo/passenger area provides adequate environmental control capacity and, even with one unit failed or shut down, will provide an adequate capacity over a majority of the airplane operational envelope. This permits keeping the airplane in operation, if required, until convenient to perform maintenance.

A crossover permits the cargo ECS units to provide conditioned air to the crew compartment in case of failure of the crew pack.

The baseline system contains other operational advantages also. For instance, in case of an engine failure where thrust is critical, the two large units could be shut down to reduce power drain on the engines while still providing conditioning for the crew, with the small pack at a negligible power drain.

Operationally, one or both cargo packs could be shut down temporarily during critical takeoff portions of the mission during severe hot day environments, to reduce performance penalties.

The use of two identical units will reduce the number of required spares, as compared to two different size units.

10.4 FLIGHT CONTROL SYSTEM

In general, all primary flight control surfaces are conventional in the high-speed or non-STOL mode and consist of ailerons, elevators/stabilizer, and a rudder. All are fully powered by four completely independent hydraulic systems and are irreversible.

To provide adequate control power in the STOL mode, the elevators and rudder are doubled hinged and the ailerons are coupled to the spoilers. For preliminary weight and trade study analysis, all control systems were assumed to be of the cable and pully type with local pushrod torque shaft assemblies. All control servos were sized to allow for the loss of one hydraulic system

without maneuver capability reduction or loss of two hydraulic systems with reduced maneuver capability, and include provisions for dual electrical automatic control inputs with appropriate feedback signals. Necessary control system variations from this overall description are discussed in their respective sections.

10.4.1 PRIMARY FLIGHT CONTROLS

10.4.1.1 Pitch Control System

Pitch control is provided to the pilots by the use of center-mounted control columns, one for each pilot. Identical yoke-type grips are provided, with the switches and buttons required to accomplish the necessary functions. Flight control switch functions include DLC, trim, and autopilot disengage. Column pitch displacement is transmitted to the control system by means of pushrods and a single-loop cable system to control a pitch master cylinder. This mechanical system is attached to the pilot's and copilot's columns with a disconnect mechanism. Located at the copilot's column are the position transducers which provide an electrical control path through pitch SCAS. Artificial feel is provided by feel bungees, one at the master cylinder on the pilot control linkage and one at the copilot's column, each providing part of the feel gradient under normal circumstances. The disconnect protects against a jam in the control system to, and including, the master cylinder. Disconnecting the columns allows the pilot to fly the air vehicle using the mechanical path should the copilot's wheel jam, or allows the copilot to fly the air vehicle using SCAS should the pilot's wheel become jammed. The pitch master cylinder is used to isolate the SCAS from pilot displacement inputs. A system irreversible from pilot force is thus provided, reducing friction and required force levels and preventing feedback from augmentation control commands. The output of the pitch master cylinder is position summed with SCAS by means of dual path walking beam linkages. Dual path linkages are used to provide continued operation for a single disconnect-type failure. The summed output is transmitted to the elevator actuators by means of push rods.

10.4.1.1.1 Stability and Control Augmentation - Pitch

The pitch SCAS transforms pitch column displacement and air vehicle motion about the pitch axis into elevator deflection by means of electrical signals to the pitch servos and mechanically from the servos to the elevator actuators. The displacement signals from sensors located at the copilot's column are gain scheduled, as required, and control servo displacement proportionately to column displacement. Air vehicle motion is sensed about the pitch axis by rate gyros and normal accelerometers whose signals are shaped and gain scheduled, as required, to control servo displacement proportionate to pitch rate and normal acceleration, respectively. SCAS thus transforms

these signals from pilot inputs and air vehicle motion into elevator deflection to produce the desired damping and maneuver control.

The pitch SCAS is a dual-dual configuration to provide flight safety in a fail-operational system. Two independent dual-channel systems are mechanized with dual-tandem servos which are position summed on walking beam linkage. Each dual-tandem servo is powered by a single hydraulic system for rigid control of failures and to easily provide balancing for interchannel deviations during normal operation. A monitor automatically disengages the individual channel if an unacceptable interchannel difference exist for longer than a specified period of time. The summed walking beam displacement, as well as individual servo displacement, is fed back to the servos in a ratio which provides walking beam displacement to servo input, essentially constant for one servo operating or both servos operating, as the means of supplying fail-operational characteristics.

10.4.1.1.2 Pitch Trim

Pitch trim is provided by two electromechanical trim actuators position-summed with the pitch master cylinder outputs on walking beam linkage. Pilot trim command is by actuation of a trim switch located on each respective yoke.

The normal mode of operation includes speed stability and trim for take-off. A two-motor actuator is used to allow monitoring between channels in the normal mode for automatic disengage after failure.

An alternate mode of operation is available with pilot selection on the center console, trim power switch. The alternate mode provides rate trim only, from the yoke grip switches.

10.4.1.2 Roll Control System

Roll control is provided to the pilots by the use of the center-mounted yoke-type control wheels. Wheel displacement is transmitted to the control system by means of push rods and a single-loop cable system to control a roll master cylinder. This mechanical system is attached to the pilot's wheel, which connects to the copilot's wheel through a disconnect mechanism. Located at the copilot's wheel are position transducers which provide an electrical control path through roll SCAS and the electrical aileron/spoiler control.

Artificial feel is provided by feel bungees, one at the master cylinder on the pilot control linkage and one at the copilot's wheel, each providing one-half the feel gradient under normal circumstances. The disconnect protects against a jam in the control system to, and including, the master cylinder. Disconnecting allows the pilot to fly the air vehicle using the mechanical path should the copilot's wheel jam, or allows the copilot to fly the air

vehicle using the electrical aileron/spoiler control and SCAS should the pilot's wheel become jammed.

A roll master cylinder is used to isolate the aileron/spoiler control mechanism from pilot displacement inputs. A system irreversible from pilot force is thus provided, reducing friction and required force levels for pilot control. The output of the roll master cylinder is transmitted through non-linear linkage and dead band to the aileron actuators or aileron/spoiler actuators, with required phasing and dead band removed for STOL operation.

10.4.1.2.1 Stability and Control Augmentation SCAS - Roll

The roll SCAS transforms wheel displacement and air vehicle motion about the roll axis into aileron deflection by means of electrical signals to the aileron servos. Air vehicle motion is sensed about the roll axis by rate gyros which have signals that control servo displacement proportional to roll rate. The roll SCAS servo configuration is identical to pitch SCAS.

10.4.1.2.2 Roll Trim

Roll trim is provided through the use of the aileron servos with control switches located on the center console. A high degree of flexibility in trim rate and authority is thus provided for a minimum amount of hardware.

10.4.1.3 Yaw Control System

Yaw control is provided to the pilots by use of sets of pedals for each pilot. Each pedal set is adjustable individually for pilot leg length. Pedal displacement is transmitted to the upper rudder control systems by menas of push rods and a single-loop cable system. This mechanical system is attached to the pilot's pedal linkages, which permanently join the pilot and copilot podal sets. Also attached at the copilot's pedal linkages are the position transducers, which provide an electrical control path through yaw SCAS to control the rudder. A system irreversible from pilot force is thus provided, preventing pedal feedback from augmentation control commands. The output of the cable loop is transmitted through a rudder limiter to provide effective rudder travel limiting for high-speed flight. This linkage controls the valves on the actuators to provide rudder displacement proportional to rudder pedal position.

10.4.1.3.1 Stability and Control Augmentation SCAS - Yaw

The yaw SCAS transforms rudder pedal displacement and air vehicle motion about the yaw axis into rudder displacement by means of electrical signals to the yaw servos, and mechanically from the servos to the rudder actuators. The

pedal displacement signals from sensors located on the copilot's pedals provide servo displacement proportionately to pedal displacement. Air vehicle motion is sensed about the yaw axis by rate gyros and accelerometers whose signals are shaped, gain scheduled, and limited, as required, and used to control servo displacement. SCAS thus transforms these signals from pilot inputs and air vehicle motion into rudder motion to produce the desired damping and maneuver control characteristics.

The yaw SCAS is a dual-dual redundant system consisting of two operating channels. Each operative channel is made up of one model and one active channel controlling an electrohydraulic servo. A monitor automatically disengages any operative channel which has active and model channels that are different by an unacceptable amount for longer than a specified period of time. The servo outputs are position summed on walking beam linkage. The summed walking beam displacement, as well as individual servo displacement, is fed back to the servos in a ratio which provides walking beam displacement to servo input, essentially constant for one servo operating or both servos operating, as the means of providing fail-operational characteristics.

10.4.1.3.2 Yaw Trim

Yaw trim is provided by two electromechanical trim actuators position summed on dual linkages which displace the yaw feel bungee and thus determine pedal displacement for zero force. The trim actuators operate from yaw trim switches located, respectively, on the pilot's and copilot's left consoles. One trim actuator is controlled at any given time with the second unit in a standby condition. A yaw trim power switch on the pilot's center console provides the means to select which of the actuators is operating; i.e., normal or alternate.

10.4.2 SECONDARY FLIGHT CONTROLS

10.4.2.1 Flaps and Slats

The flaps are mounted in three-per-side sections. Extension or retraction is accomplished by dual hydraulic motors driving jackscrew actuators through a conventional torque tube system. An asymmetry detector, located at the outboard end of each torque tube, locks the motor shaft through a disk-type position brake if an unacceptable degree of asymmetry is reached. The aft segments of the inboard and outboard flaperon sections form integral parts of the direct drag control and roll control systems.

Twelve (six per wing) leading-edge slat panels extend across the span with necessary clearance sizing and breaks. Each section is hydraulically

extended to its optimum deflection upon selection of any flap setting, before flap motion is allowed. The hydraulic actuators are equipped with an asymmetry detector/brake system.

10.4.2.2 Spoilers

The eight spoilers (four per wing) serve several functions. Simultaneous operation can only occur in flight when the flaps are fully retracted for use as a speed brake, or on the ground when a microswitch on the nose gear strut is closed for wing-lift reduction. Sequential operation for additional roll control occurs in the STOL mode (nonzero flap setting with the microswitch open). Singular operation of the inboard panels for direct-lift control requirements occurs only when the flaps are in their full or landing position.

Separate duel hydraulic servo actuators enable the required differential and symmetrical deployments. Electrical provisions are incorporated for automatic and pilot-commanded control on the inboard panels.

10.4.2.3 Direct Lift and Drag Control

Two types of control surfaces are utilized in this system which control drag as well as lift. Inboard spoiler panels are employed to control lift, while the aft segment on the inboard flap sections are used for drag control.

Coincidental with landing flap positioning, each pair of control surfaces is symmetrically rigged by dual electrically controlled hydraulic irreversible servos to a predetermined setting which provides the lift-to-drag ratio required at design approach speed and power setting. As presently conceived, drag modulation is commanded through a switch on the control column, and the lift command signal from a trim-type button is superimposed with a signal from the roll control system as it exceeds a preset dead band. Space, weight, and power allotments are made for additional automatic or augmentation controls.

10.4.2.4 STOL Mode Conversion

Two pilot-controlled actions are necessary to convert from conventional flight to the STOL mode, or vice versa. These actions are flap selection and diverter-nozzle positioning.

Activation of the flap selection lever from the retracted position results in the following:

1. Releases null-position hydraulic locks on both elevator and rudder forward segment actuators, allowing these surfaces to be coupled into their respective control systems.

2. Closes the circuitry to the sequential spoilers, which provides the roll control system with maximum capability.

3. If landing flaps are selected, activates the direct lift control system.

Positioning the diverter-nozzles concurrently resets the horizontal stabilizer through the pitch trim system. The command signal can either be a predetermined value or calculated by an air data computer if it is necessary to increase the STOL mode operational range.

10.5 FUEL SYSTEM

Fuel is contained in two interconnected wing tanks in each wing and in four separate main tanks located in the center wing section, above the cargo compartment. A schematic diagram of the fuel system is included in figure 117.

Consideration given to methods of preventing or limiting fuel loss due to damage from small-arms projectiles includes the following:

1. The four main tanks are strategically located to take maximum advantage of available structural protection.

2. An additional safety feature is incorporated in the form of self-sealing material applied to the bottom and sidewall surfaces of the main tanks.

3. Fuel leakage from the integral portion of the tank, outboard of the engines pods, will not promote a fire because fuel will normally be used from the outboard wing tanks first.

Use of the composite sealing system developed by the Air Force Material Laboratory offers a number of advantages, in this application, over the self-sealing cell method. The overall weight can be held to a minimum through application of the self-sealing material to only those areas of the fuel tanks that require projectile inflicted leakage control. Smaller and fewer access openings are required, because the composite sealing materials may be applied to external tank surfaces as well as to internal tank surfaces.

Fire and explosion suppression within the fuel tanks is accomplished through the use of a nitrogen inerting system. Dilution to less than 10-percent oxygen content in the ullage space prevents explosion of fire. For

this application, nitrogen inerting is considered superior to other suppression techniques such as reticulated foam. The nitrogen inerting system advantages include a significant weight savings, improved utilization of fuel tank volume, and enhancement of access for maintainability.

10.5.1 FUEL TRANSFER AND ENGINE FEED

Fuel contained in the outboard wing fuel tanks flows by gravity to the inboard wing tanks and is pumped to the inboard wing transfer pump manifold for delivery to all four main tanks. Fuel transfer into each main tank through normally open, dual level control valves continues until all fuel in the wing tanks is depleted, at which time transfer pumps are automatically shut off.

Each engine is fed from a separate, independent main tank containing one fuel booster pump. Any one main tank can also supply fuel to any combination of two, three, or four engines when the applicable interconnecting crossfeed valves are opened. Suction feed from each main tank is provided for use in case of booster pump failure and/or until fuel feel from another main tank can be established. Fuel supply to any engine may be shut off by closing the applicable firewall shutoff valve.

10.5.2 REFUELING/DEFUELING AND AERIAL REFUELING

Two pressure refueling receptacles are located on the side of the airplane, as low as possible, to provide easy access from the ground without the need for workstands. The required refueling flow rate of 600 gallons per minute per receptacle is supplied to the fuel system interconnecting lines through dual level control valves in each tank. The servicing panel includes override switches to test each stage of each dual level control valve for proper fuel shutoff at the start of refueling.

Defueling is accomplished through the pressure refueling receptacles of 200 gpm. By opening the refueling line shutoff valve and closing each of the main tank level control valves, fuel may be pumped directly to the receptacles from the wing tanks. Any one or all of the main tanks may be defueled by operating the booster pump(s) with the refueling shutoff valve closed, the defueling valve open, and the applicable crossfeed valve(s) open. Gravity filling and/or suction defueling openings are provided in each tank for emergency use.

The universal aerial refueling receptacle slipway installation, located on the top centerline of the airplane and aft of the crew compartment, is connected by a check valve to the same fuel distribution system used during ground refueling. After ground or aerial refueling, fuel trapped in the refueling line is pumped to one of the main tanks through the scavenge system.

The universal aerial refueling receptacle slipway installation, located on the top centerline of the airplane and aft of the crew compartment, is connected by a check valve to the same fuel distribution system used during ground refueling. After ground or aerial refueling, fuel trapped in the refueling line is pumped to one of the main tanks through the scavenge system.

10.5.3 VENTING AND NITROGEN INERTING

Dive and climb vent outlets are provided in each tank, with interconnecting lines manifolded to one common vent valve. Regulated nitrogen gas, supplied from the liquid nitrogen dewar, provides a positive pressure differential throughout the fuel system and fills the ullage space to produce an inert atmosphere as the fuel is used. The nitrogen pressure also services the vent valve servos as a reference pressure to maintain a closed vent system until an excess positive differential pressure actuates the valve open. The vent valve automatically goes to the open position if nitrogen pressure is depleted. Emergency dive and climb vent relief valves are also provided in case the vent valve fails closed.

10.5.4 FUEL DUMP

A fuel dump shutoff valve is located outboard of each wing tank and receives fuel to be dumped from the wing tank transfer pumps. During the fuel dump process, fuel may be also transferred to the main tanks or may be restricted from entering the main tanks by closing the level control valves. The fuel in the main tanks cannot be dumped in order to insure an adequate reserve for engine operation.

10.5.5 FUEL QUANTITY GAGING

The fuel gaging system incorporates liquid-level-sending capacitor-type probes installed in each tank. Total fuel quantity readout is provided, together with individual fuel tank quantity displays. At the normal flight attitude, the fuel quantity gaging system is designed to provide readout accuracies within the range of ±0.75 percent of full scale ±2 percent of fuel remaining, as required by MIL-6-7940, class II, accuracy.

10.6 LANDING GEAR

The landing and deceleration systems include the main and nose landing gears; fairing doors complete with actuating and locking systems, nosewheel steering, and damping system; wheels, tires, and brakes; brake control and antiskid system; and all operational mechanisms.

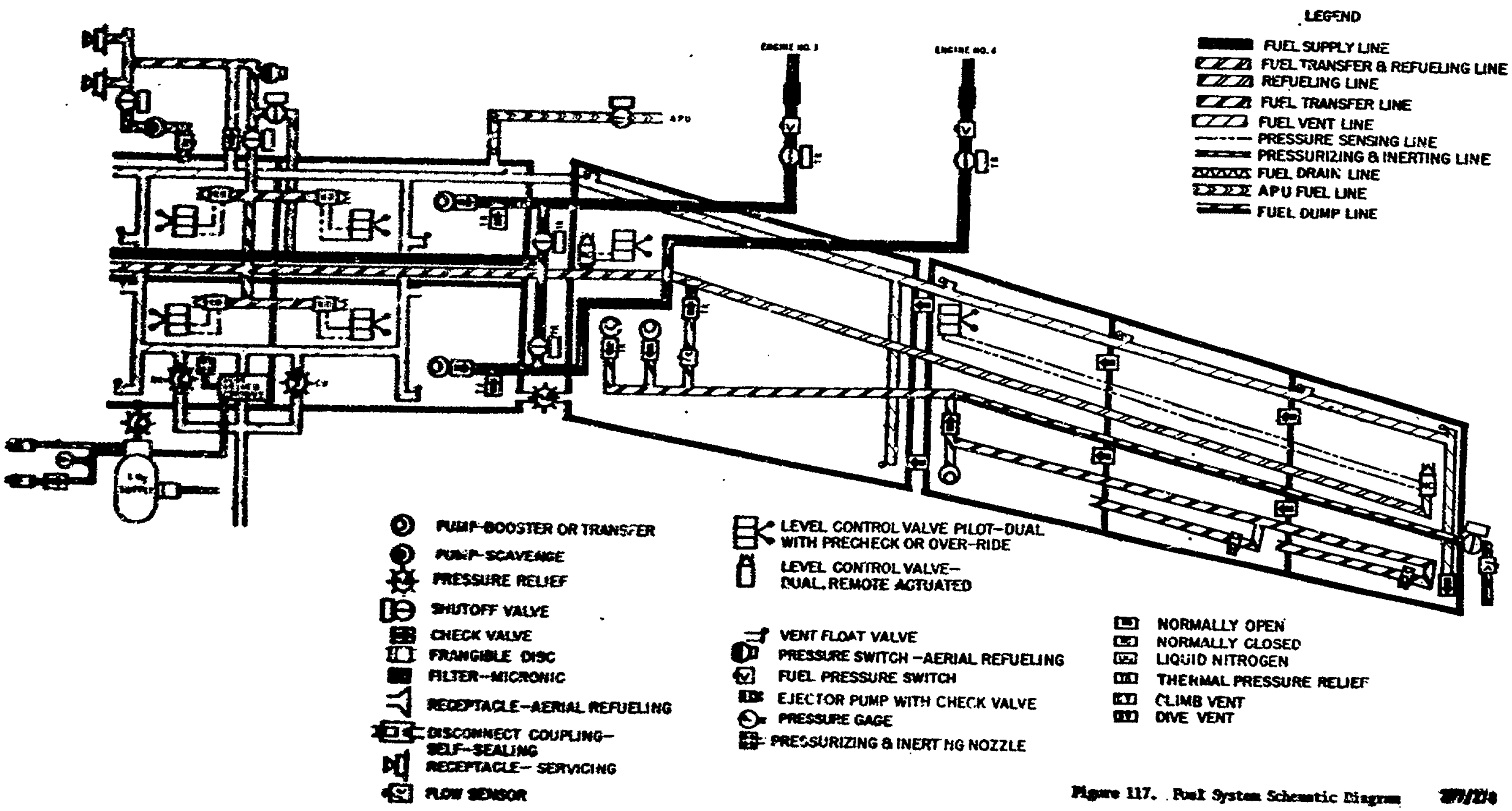


Figure 117. Fuel System Schematic Diagram

The landing gear system has been designed to meet the desired STOL landing and flotation objectives of CBR 6 and 200 passes and a limit sink speed of 15 feet per second at midmission weight. It has been designed to the provisions of MIL-A-008862A, paragraph 3.2.7. The standard ultimate load factor of 1.5 applies to the landing gear and its supporting structure. The gear is designed with sufficient tread width and strength to allow good crosswind operation, and meets the turnover requirements of USAF Design Handbook DH2-1. The aircraft is equipped with an antiskid, automatic actuating braking system. The landing gear is capable of a one-time STOL takeoff and return to a main operating base after one nose or one main gear tire has been punctured.

The landing gears are electrically controlled from the crew station and are hydraulically operated to achieve minimum weight and space requirements. Each of the main gears and the nose gear has redundancy for operation to a safe landing configuration, in the event of electrical or hydraulic failure. Both main and nose gears are forward retracting to provide a free-fall capability once the uplocks are released.

The main gear and the nose gear are fitted with auxillary actuators to unlock the doors and the uplocks, to allow the gears to extend for emergency operation. All uplocks are provided with redundant release mechanisms, and are held overcenter and locked in the desired closed or open position, with or without hydraulic power.

The configuration and arrangement provide compatibility between aerodynamic design and landing gear structural and dynamic requirements. The geometric locations of the wheels, in relation to the aircraft center of gravity, are set primarily to obtain satisfactory tip-back angle and lateral stability characteristics. The nose gear location has been optimized, considering the factors of available stowage volume and forward-folding kinematic arrangement.

A four-wheel bogie, twin-tandem arrangement mounted on a single strut has been selected for the main gears; a twin-wheel, single-strut arrangement has been selected for the nose gear. Location and sizing of the wheels and tires meet requirements of AFFDL-TM-71-09-FEM, "Design Procedure for Establishing Aircraft Capability to Operate on Soil Surfaces," dated 6 September 1971. All gears are enclosed by contoured fairing doors during flight, and the nose wheel doors are closed during taxi, takeoff, and landing. Tires selected are a new design interpolated from existing type III tire data. The landing gear subsystem provides the following functions:

1. Absorbs and/or transmits the static and dynamic energy resulting from the air vehicle landing and ground maneuvering operations.

2. Provides directional control of the steering, turning, pivoting, taxiing, braking, and damping of the air vehicle.

3. Provides ground flotation for the air vehicle during ground maneuvers.

4. Provides for retraction, extension, and locking of the gears for the flight and ground modes of operation, and indication and warning to the pilot of gear position.

5. Provides for towing, tiedown, holdback, and jacking of the air vehicle.

Figure 118 indicates the landing gear general arrangement and turning capability applicable to the STOL TAI airplane. The nose gear can be turned to an angle of 77° 15' to permit a 180-degree turn of the aircraft on a 60-foot-wide strip.

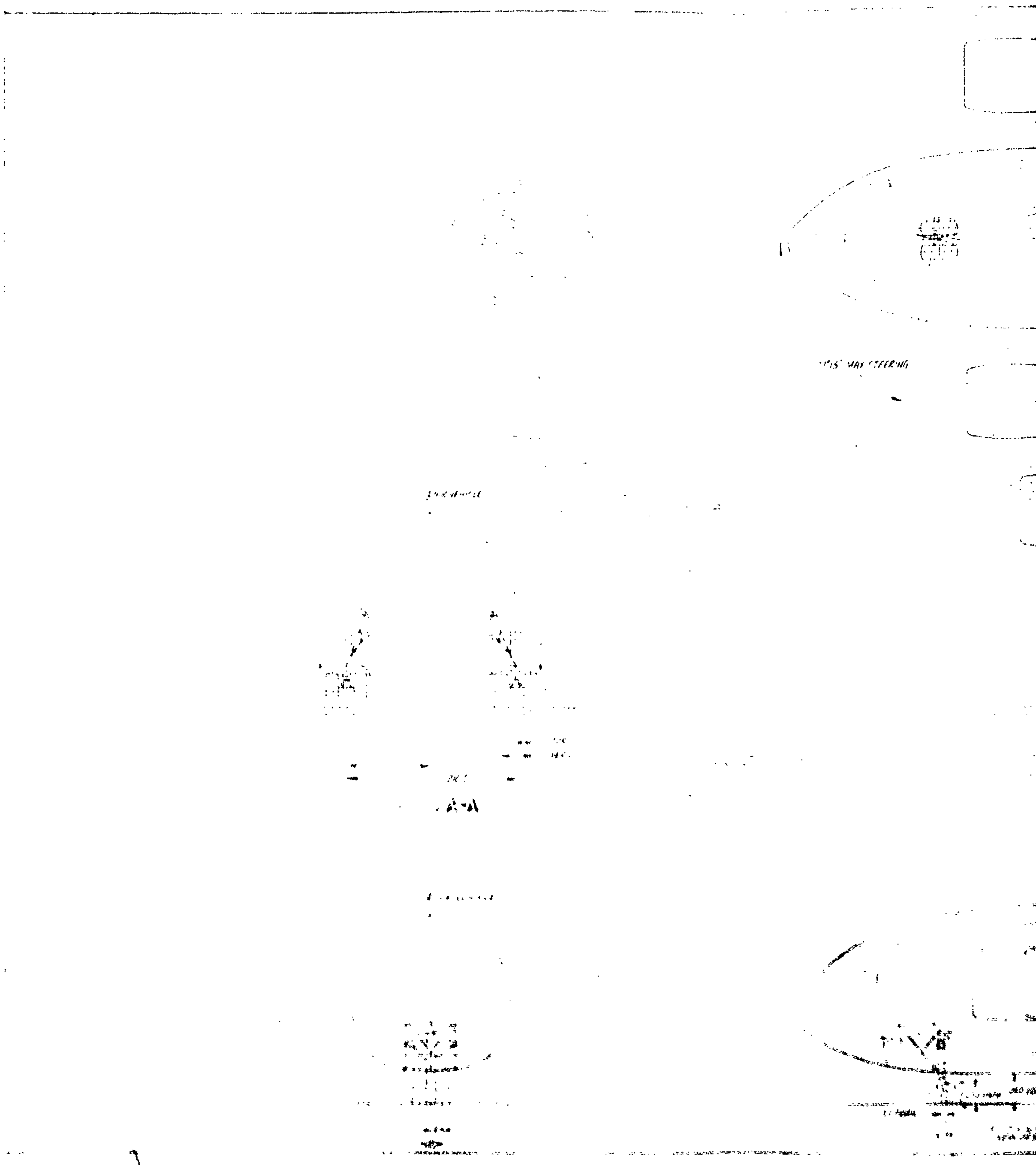
10.7 MATERIAL HANDLING

The material handling system includes provisions to accommodate USAF 463L material handling system-type unitized cargo. The 463L cargo restraint provisions are integral with the cargo bay floor for structural efficiency and durability and to facilitate reconfiguration to alternate mission modes. Provisions are included to accommodate intermodal (8- by 8-by 10-foot multiples) containers. Provisions are included to accommodate a powered-pallet transfer device for one-man on/off loading of palletized cargo, in addition to the conventional winching capability for movement of both palletized and vehicular cargo. One-man pallet positioning can be accomplished using a portable electrically powered (28-volt dc) transfer device which fits under the pallets and reacts against a rack-gear surface submerged in the floor.

The cargo restraint provisions in the cargo floor will accept the additional aerial delivery system (ADS) peculiar restraint and release mechanisms. Space and mounting provisions are available in the cargo bay overhead, just forward of the pressure door retracted position, for installation of ADS extraction parachute release mechanism. Control systems for both normal (cockpit) and secondary (cargo bay) actuation of the ADS release mechanism are integral portions of the aircraft.

The indicated cargo compartment dimensions represent unobstructed net usable space available. Ample space is provided in the aircraft sidewall areas for stowage of all material handling system functional hardware not in use, without intruding on the usable cargo space.

The specified payload is net of payload-accommodation equipment (furnishings) such as cargo tiedowns, troop seating/comfort/survival provisions, and litter accommodations. The weight allowances for these items have been included in the OWE totals. Thus, the specified payload is entirely net


WAY STEERING

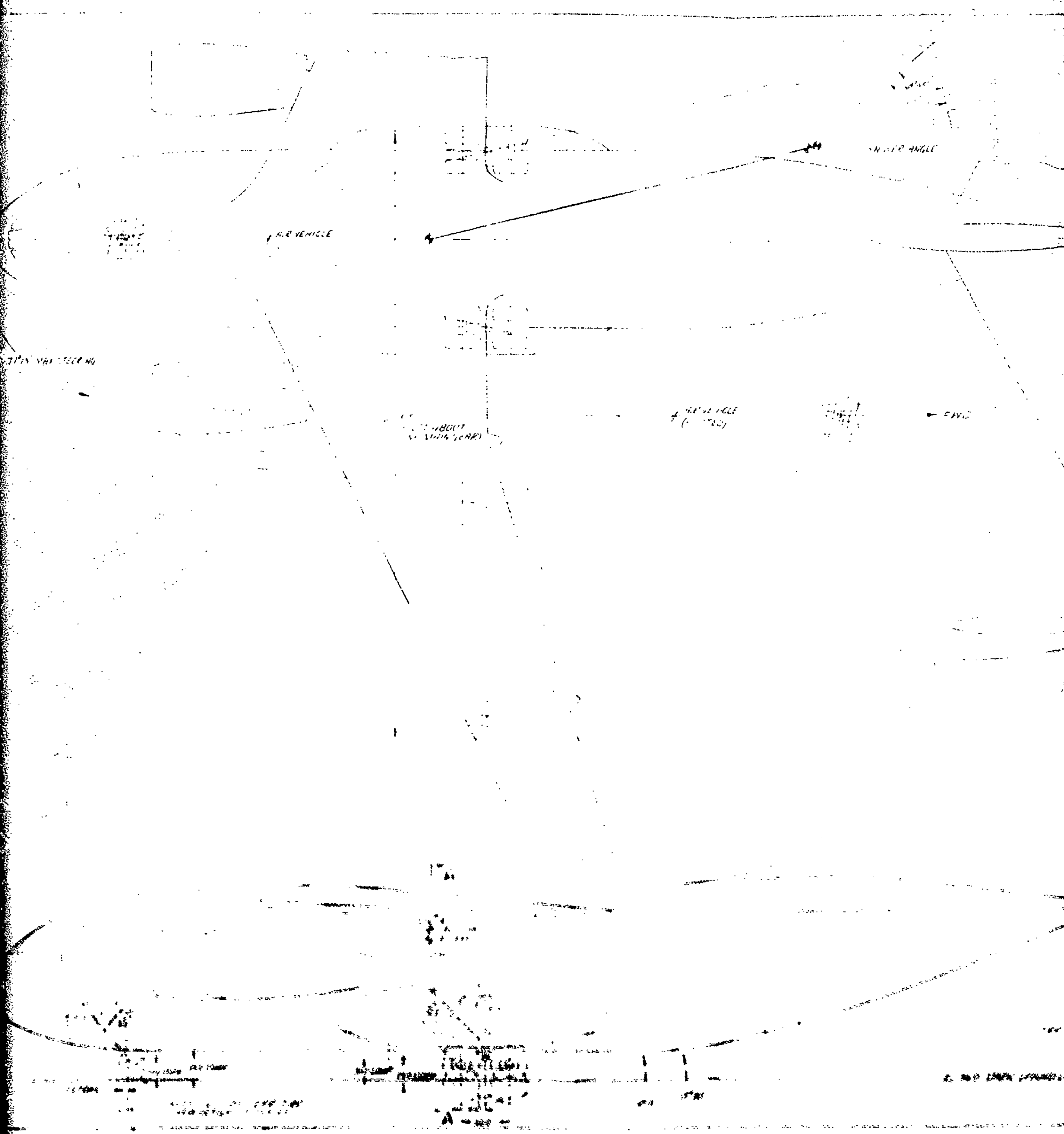


Figure 118.

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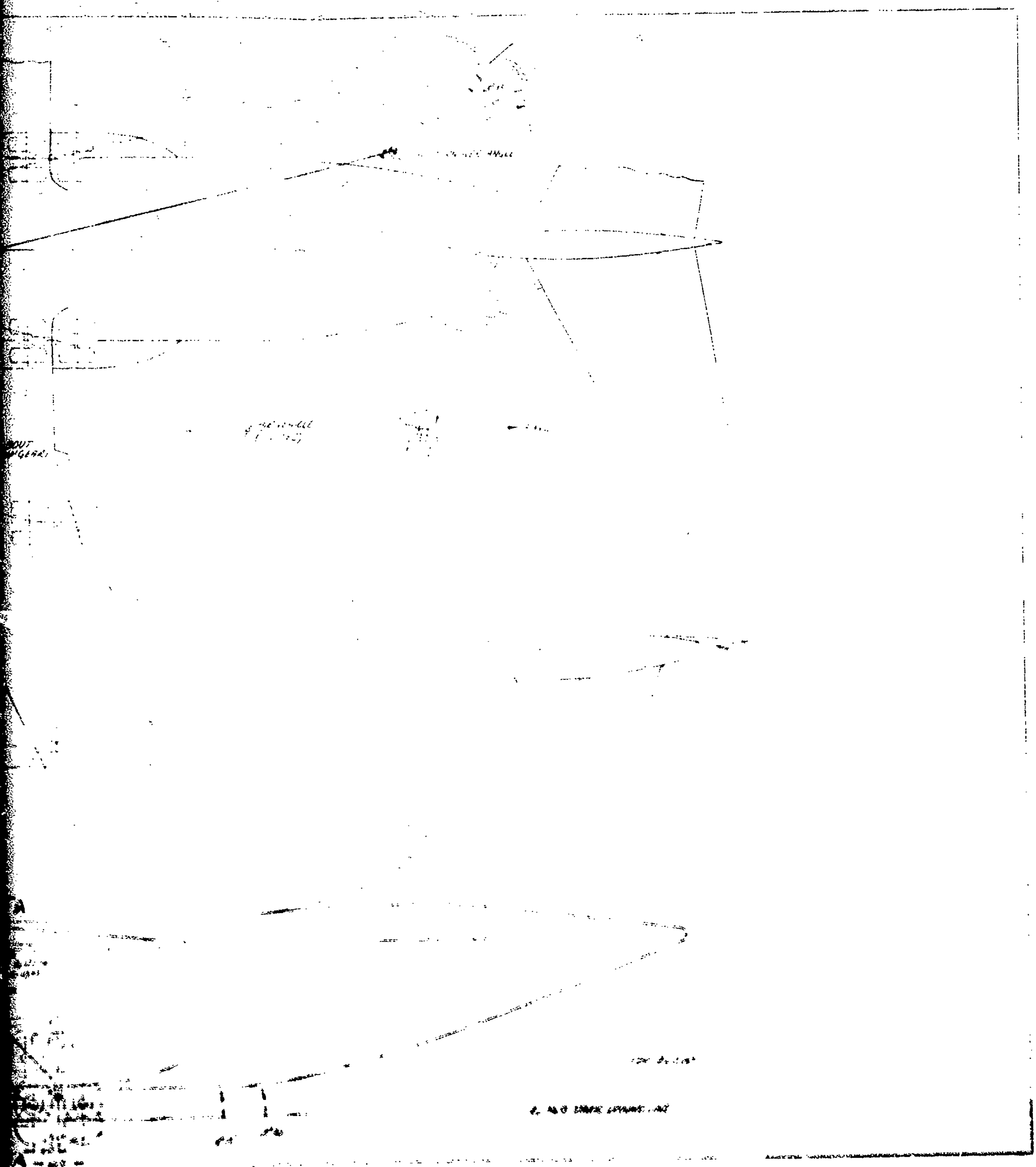


Figure 118. Landing Gear Concept

useful load. The STOL TAI is able to deliver rated payloads while retaining, at any sortie completion, a self-contained capability for reconfiguration to different mission types.

When utilized for palletized logistics cargo, maximum density is attained with the 88- by 108-inch pallets, carried with the 108-inch dimension oriented laterally in the aircraft. The cargo floor width of 140+ inches leaves a floor level scanning aisle, along each side, of approximately 18 inches, with pallets loaded to a 104-inch-width limit.

As shown in figure 119, this provides ample room for personnel movement to check the security of the palletized cargo, as well as the alternate mission equipment stowed along the wall. For the ADS mission, using 108-inch-wide pallets, the internal clearances are the same.

When utilized for paratroop airdrop missions, the cargo bay would be configured as depicted in figure 120. This four-row arrangement provides wide egress aisles and, based upon 24-inch seat width, will carry 74 paratroops. For troop ferry missions, a smiliar four-row arrangement will provide a capacity of 90 20-inch seats. If desired, an additional five seats per side can be mounted on the wheel well walls, giving a total of 100 seats. An alternate five-row, high-density configuration can provide a capacity of 116 20-inch seats, as depicted in figure 121. Figure 122 depicts the cargo bay floor plan for the several troop-carrying configurations previously noted.

10.8 SECONDARY POWER

An auxiliary power unit (APU) is required onboard the vehicle to provide electrical power, hydraulic power, and pressurized airflow for the environmental control system during ground and emergency conditions when the main propulsion system is not in operation. In addition to these functions, the APU will be used to start the main engines by means of a pneumatic starting system. The pneumatic starting system was selected from other methods such as electrical, cartridge, fuel-air, hydraulic, and cryogenics, based on reliability, weight, cost, and risk. The size of the APU gas turbine is established by the amount of pressurized airflow available to properly start the main engines.

A 100-percent size AiResearch GT CP 165 series unit, used in the B-1 bomber and C5-A transport, would satisfy the engine starting and output shaft horsepower requirements fot the STOL TAI vehicle, which uses the 93-percent size GE 13/F10 turbofans. The compressor section is designed to provide compressed air required for the engine-turnine operation, as well as the large quantity that is taken as pneumatic power for external use. The accessory section includes an integral output pad for driven equipment, as well as the necessary electrical, fuel, and oil system components. The unit is equipped

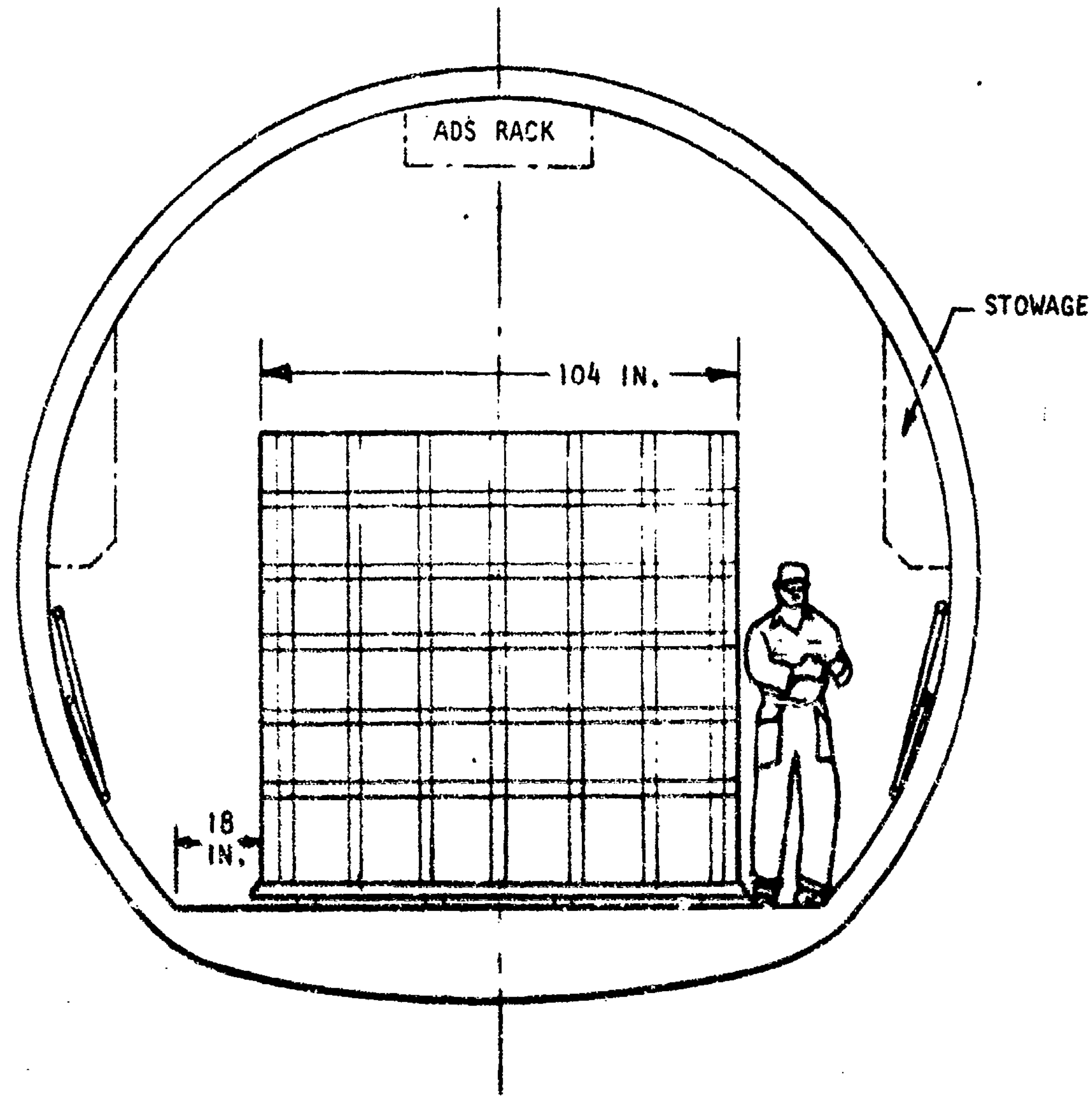
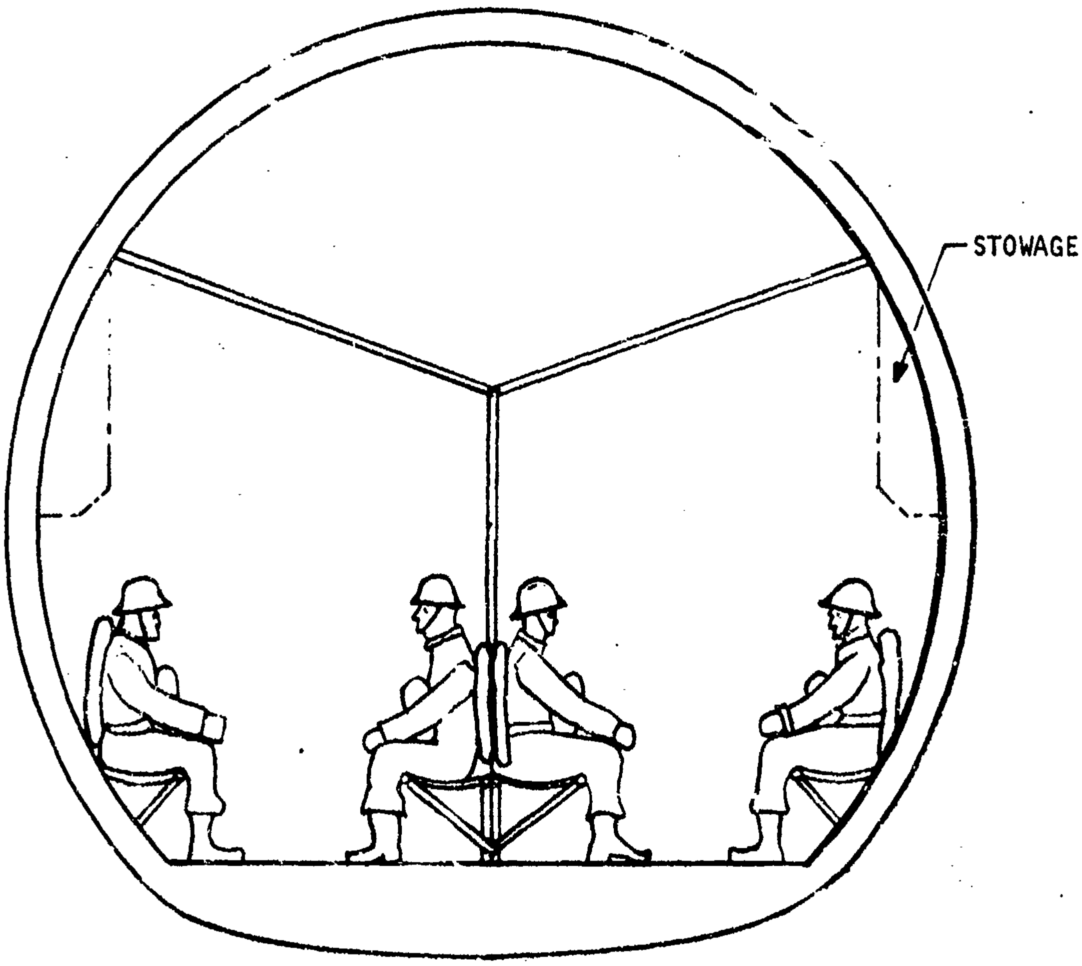


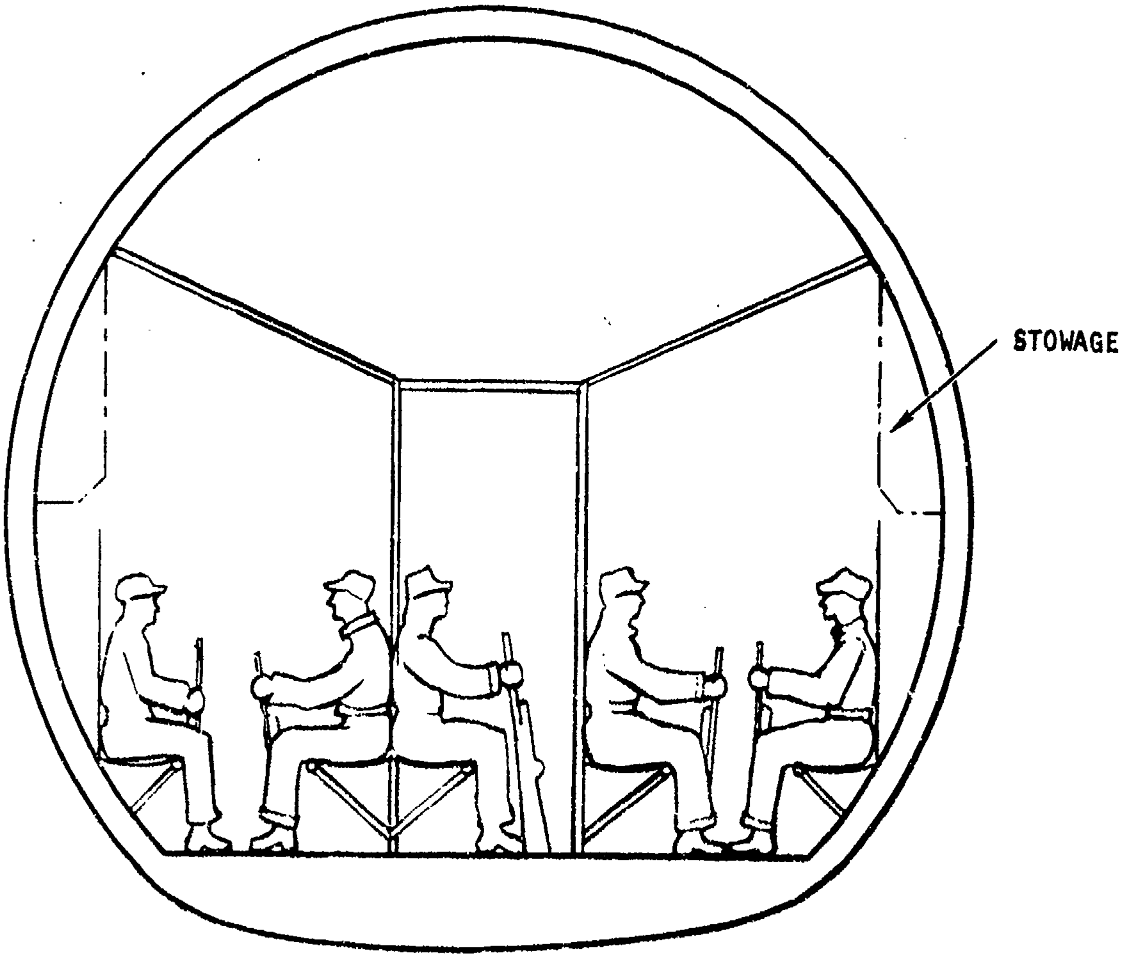
Figure 119. Cargo Bay Configuration - Palletized Cargo Mission



CARGO BAY CONFIGURATION
PARATROOP AIRDROP MISSION
4 ROWS, 24 IN. SEATS
CAPACITY: 74 MEN

LOW-DENSITY TROOP FERRY MISSION
4 ROWS, 20 IN. SEATS
CAPACITY: 90 MEN

Figure 120. Cargo Bay Configuration - Troop Mission (Four-Row)

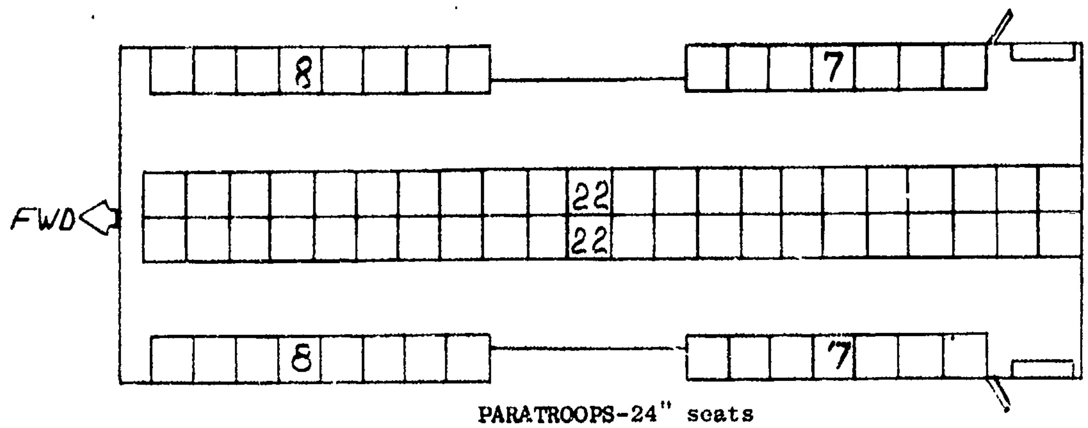


HIGH-DENSITY TROOP FERRY MISSION

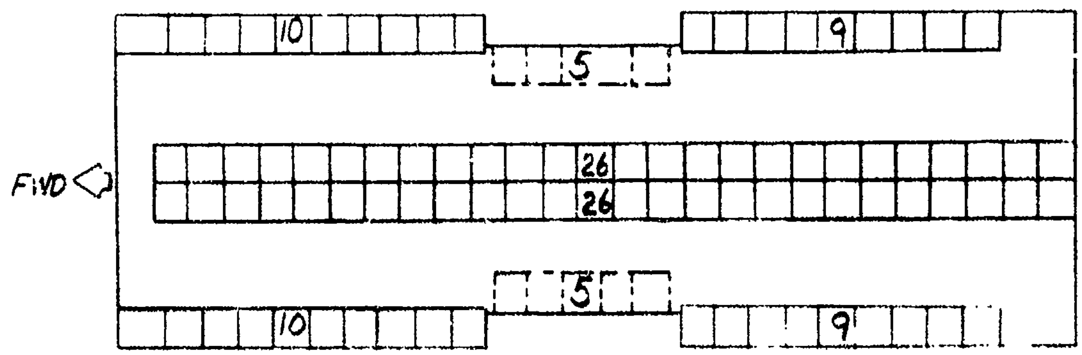
5 ROWS, 20 IN. SEATS
CAPACITY: 116 MEN

Figure 121. Cargo Bay Configuration - Troop Mission (Five-Row)

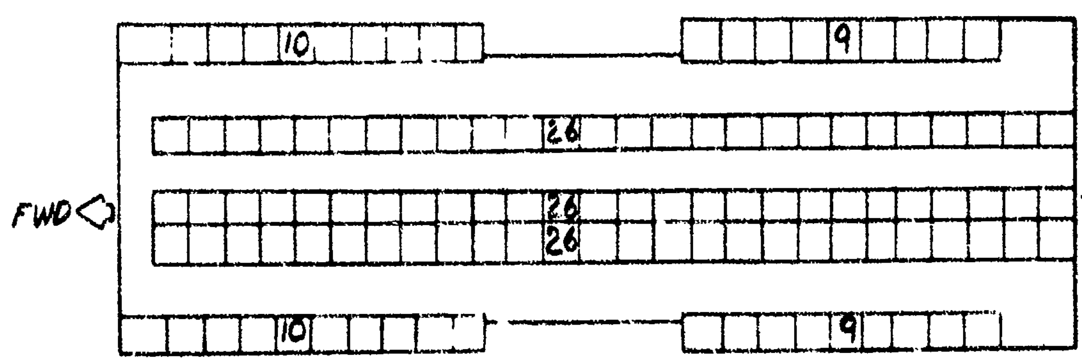
CARGO BAY SEATING ARRANGEMENT
(for 45 ft. x 11.7 ft. cargo bay)



PARATROOPS-24" seats
Capacity 74 men



TROOP FERRY-Low Density-4 Rows
20" seats-Basic capacity 90 men; optional capacity 100 men



TROOP FERRY-High Density-5 rows
30" seats-capacity 116 men

Figure 122. Cargo Bay Floor Plan

with a fully automatic control system that properly sequences starter operation, fuel control, ignition, and surge control to provide a stable automatic starting cycle requiring only the engagement of a single starter switch. Additional automatic controls provide continuous operating adjustments to compensate for variations in bleed airload, shaft power requirements, and ambient conditions.

The APU is installed with a self-contained hydraulic starting system that can be used for ground or emergency flight operation. Two accumulators installed in separate hydraulic systems provide a double-start capability without an additional power source. The accumulators are charged when the hydraulic systems are pressurized by operating the pumps or by supplying external pressure. A battery is required onboard to provide electrical energy for opening the fuel flow valve, although the valve would also have a manual override control.

10.8.1 ELECTRICAL SYSTEM

The aircraft electrical system provides 115/200-volt, three-phase 400-hertz ac and 28-volt dc power to the load equipment. The main ac system is a variable-speed constant frequency (VSCF) system with four 40-kilovolt-ampere (kva) oil-cooled generators, four power frequency converters, automatic synchronizing/paralleling equipment, four distribution buses, and supervisory and protective equipment. One generator is driven by each of the aircraft engines. An APU drives a 20 kva generator for use during an in-flight emergency or on the ground when ground power is not available. The dc system consists of two 100-ampere transformer-rectifier units (TRU); a 28-volt, 36-ampere-hour battery; two main dc buses; and a battery bus. An external receptacle is provided for ac ground power for aircraft servicing and maintenance. A schematic diagram is shown in figure 123, and a load analysis is shown in figure 124.

The VSCF system has been chosen for this aircraft because of its inherent reliability, stability, and efficiency. Reliability of the system has been demonstrated to be superior to the mechanical-type of constant-frequency system. Generator and converter mean-time-between-failure is of the order of 20,000 hours. The frequency in the VSCF system is controlled by a stable electronic oscillator, and the frequency of the output power does not vary with the application and removal of loads or with abrupt changes in engine speed. System efficiency is 10 percent greater than that of mechanical drivers, with resulting conservation of fuel and reduction in cooling requirements. Weight of the VSCF installation (including generator, converter, filter, control panel, and feeders) compares very favorably with other systems.

An analysis of the electrical and electronic equipment required by this type of aircraft, and the mission time-line load profile, shows the total

generator capacity required to be approximately 73 kva. An installed capacity of 80 kva would satisfy this requirement, but would not provide growth potential. In order to make provision for electrical and electronic system growth and to provide redundancy, a 160 kva system is installed. A four-generator system was chosen for reasons of reliability and logistics. In the event of a generator failure, the three remaining generators are capable of supplying the entire system load. Utilization of identical engine gearbox-generator configurations simplifies logistics.

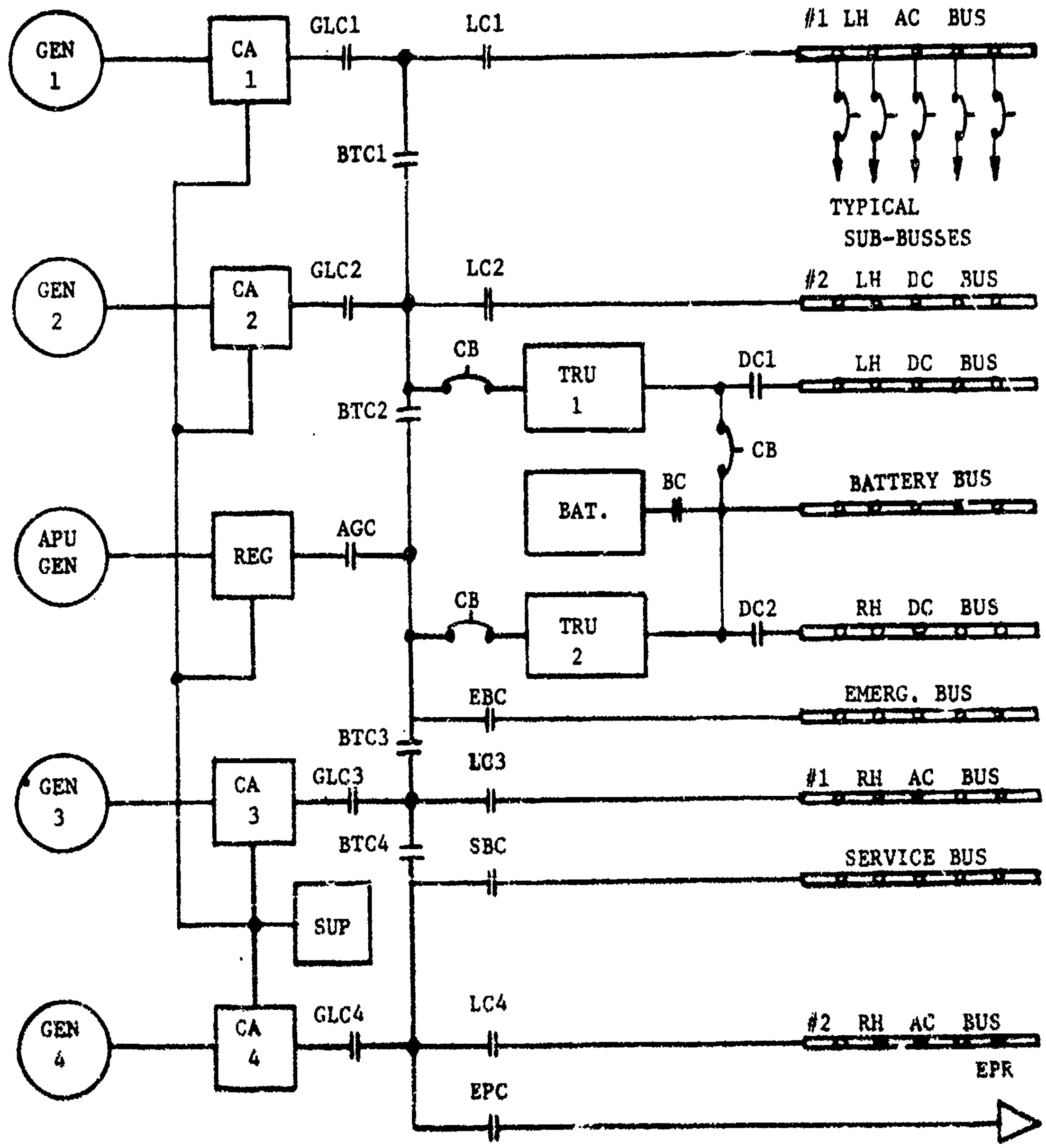
The four ac generators are used in a VSCF system and are normally operated in a split parallel configuration. The two right-hand and the two left-hand pairs are operated in parallel, with each pair powering two of the aircraft ac buses. In case of a generator failure, the remaining generators and the buses are operated in parallel. Automatic synchronizing and paralleling is provided by the converter assembly (CA) and the supervisory panel (SUP).

There are four primary ac buses (emergency and service), two on each side of the aircraft. The emergency bus can be powered from either the primary system or the APU and supplies power to flight-essential equipment. Equipment which is deemed essential to flight is powered by two buses - an emergency bus and a primary bus. The service bus is used while the aircraft is on the ground for preflight, fueling, turnaround, and cargo handling, and for providing power required by personnel during live-aboard periods. Differential current transformers are installed in the generator feeders and in the distribution buses for fault protection. This system will remove power from the bus by opening the load contactor when fault current is detected. Circuit breakers are used for the protection of individual loads.

The supervisory panel (SUP) monitors and controls the ac system automatically. Its function is to assure synchronous operation of the system, to close the bus tie contactor (BTC) in case of a generator failure, and to start the APU and monitor off the main ac buses in case of the failure of all main generators. In this case, ac power would be supplied to the emergency bus only for equipment essential to flight.

The APU furnishes pneumatic, hydraulic, and electrical power for servicing and maintenance of the aircraft. It is started by a pneumohydraulic accumulator which can be manually charged. At extremely low temperatures, when battery output is low, the APU starting valve can be operated by hand.

In the case of an in-flight emergency, with all engines or generators out of service, the APU will supply power to allow a safe landing of the aircraft. The APU is operable below 25,000 feet. If the vehicle is flying above this altitude when the emergency occurs, it will be necessary to reduce altitude to 25,000 feet before starting the APU. The APU electrical output is not paralleled with the main electrical system. The system is interlocked to prevent application of APU power while the main system is operating.



AGC	APU Gen. Contactor	EPR	External Pwr Recept.
BC	Battery Contactor	GEN	Generator
BTC	Bus Tie Contactor	GLC	Gen. Line Cont.
CA	Converter Assem.	LC	Load Contactor
CB	Circuit Breaker	REG	Regulator
DC	DC Bus Contactor	SBC	Service Bus Cont.
EBC	Emerg. Bus Contactor	SUP	Supervisory Panel
EPC	External Power Cont.	TRU	Trans. Rect. Unit

Figure 123. Electrical System Schematic Diagram

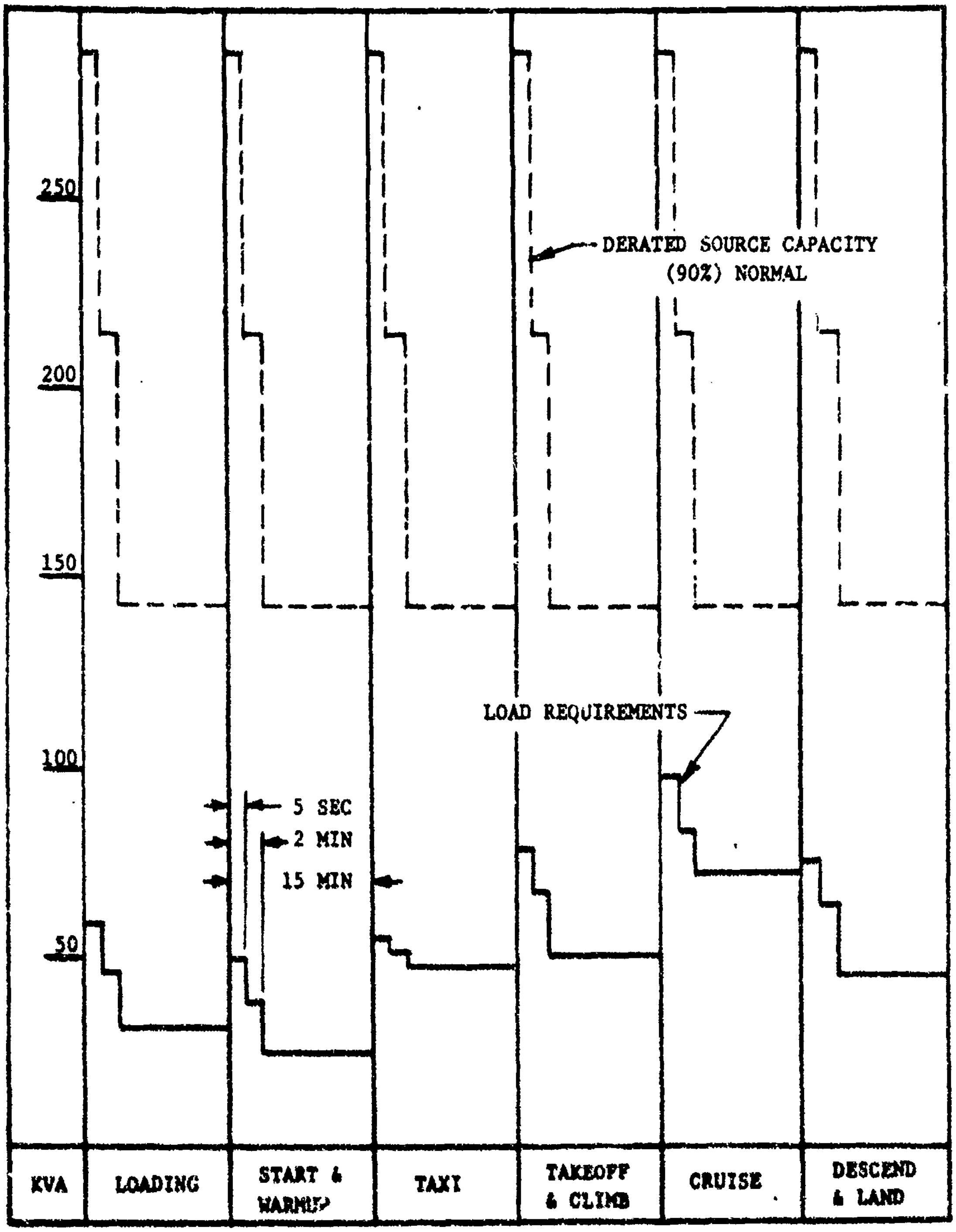


Figure 124. Electrical System Load Analysis

The dc loads are normally supplied by the two TRU's operating in parallel through the two dc buses. In case of a TRU failure, the remaining unit can supply the full aircraft dc load requirements. DC loads which are essential to flight are supplied from both buses. The battery is used for crew entry lights and APU starting. The battery contactor is interlocked with the main dc bus contactors so that no heavy dc loads can be connected to the battery. The battery is kept in a charged condition by floating across the dc bus when the TRU's are operating.

10.8.2 HYDRAULIC SYSTEM

The main hydraulic power is provided by four independent systems with a nominal working pressure of 4,000 psi and an operating temperature range of -65° to +275° F. Each system is powered by two variable displacement pumps installed on air vehicle mounted, engine-driven gearboxes. Hydraulic systems No. 1 and 2 extract power from the two left-hand engines, and systems No. 3 and 4 from the right-hand engines. The two pumps in each system are powered by different engines. A small auxiliary system, powered by the APU, is provided for ground operation of the cargo doors and ramp without running the primary flight engines.

A simplified block diagram of the hydraulic system is shown in figure 125. Two of the main systems power only the primary flight controls. The other two systems are designed for combined operation of the primary flight controls, secondary flight controls, and utility functions. This system arrangement will permit mission completion with any one system failed (after gear retraction). It also permits continued safe flight and landing capability after a second system failure. System separation is provided by routing lines on both sides of the fuselage and in the wing, by routing lines ahead of the front spar and aft of the rear spar.

Preliminary data resulted in the following flow requirements:

Function	GPM/System
Elevators	10
Spoilers	4
Ailerons	3
Landing gear	18
Brakes	15
NW steering	5

Function	GPM/System
Cargo doors	10
Aerial refueling	1
System internal leakage	2

Tentative system capacities were established by the load summaries listed in table XLVIII. The systems are sized by the descent requirement of 25 gpm when the pump speed is at 71 percent of maximum. The 25 gpm demand is the sum of the landing gear requirement plus all of the other average requirements. Although the landing gear requirement applies to system No. 2 only, the other three systems have a requirement almost as great during conventional landing. For logistics reasons, all pumps will be identical.

A preliminary selection has been made of a 0.75 cubic-inches-per-revolution pump. Assuming a volumetric efficiency of 0.92, this pump will provide a flow of 19.4 gpm at 6,500 rpm, resulting in a system capacity of 38.8 gpm.

The landing gear is normally powered by system No. 2. The landing gear has a free-fall capability and, in the event of a failure of system No. 2, the uplocks are released by system No. 3. In addition, an accumulator is provided as a backup.

The brakes are powered by systems No. 1 and 4. Emergency brakes are provided by use of the APU start accumulator.

The auxiliary motor pump system provides ground operational capability of the cargo doors when powered by the APU driven generator.

An accumulator provides emergency power for the aerial refueling receptacle. The thrust deflection nozzle is activated by pneumatic actuators, and weights for this system are included in the propulsion section.

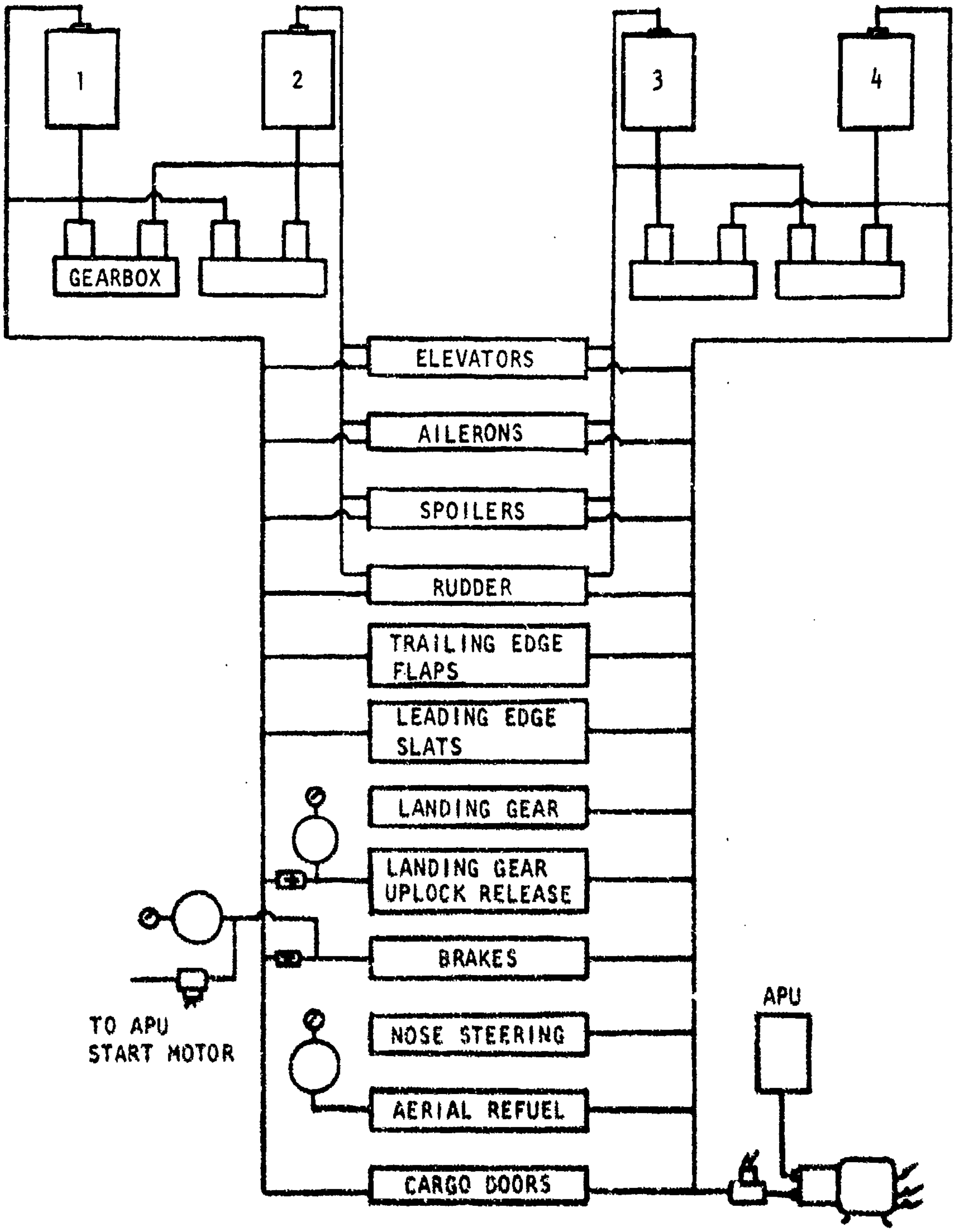


Figure 125. Hydraulic System Block Diagram

TABLE XLVIII. HYDRAULIC SYSTEM LOAD SUMMARY - GPM

	STOL Takeoff and Climb		Cruise		Cargo Airdrop		Descent		STOL Landing (flareout)		Conventional Landing (flareout)		Ground Roll	
	Avg	Peak	Avg	Peak	Avg	Peak	Avg	Peak	Avg	Peak	Avg	Peak	Avg	Peak
Elevators	5	10	1	5	1	5	2	5	5	10	2	10	1	1
Spoilers	2	4	1	2	1	2	1	2	2	4	1	4	1	1
Ailerons	1	3	1	2	1	2	1	2	1	3	1	3	1	1
Rudder	1	3	1	2	1	2	1	2	1	3	1	3	1	1
Landing gear	-	18	-	-	-	-	-	18	-	-	-	-	-	-
Brakes	-	-	-	-	-	-	-	-	-	-	-	-	3	15
Nose wheel steering	-	-	-	-	-	-	-	-	-	-	-	-	1	5
Cargo doors	-	-	-	-	-	10	-	-	-	-	-	-	-	-
Aerial refueling	-	-	-	-	-	-	-	-	-	-	-	-	-	-
System internal leakage	2	2	2	2	2	2	2	2	2	2	2	2	2	2
System average demand	14	-	6	-	6	-	7	-	14	-	7	-	9	-
System peak demand	-	32	-	12	-	16	-	25	-	28	-	22	-	21
Pump speed (% max)	100	-	90	-	90	-	-	71	-	90	-	67	-	66
Capacity required at 100%	-	32	-	13.3	-	17.8	-	35.2	-	31	-	32.8	-	31.8

SECTION XI

SURVIVABILITY-VULNERABILITY

11.1 THE STOL MISSION

The mission of the STOL transport is to deliver and sustain tactical ground forces during a conflict. The mission phases can be stated as:

- Load at secure base
- Takeoff
- Cruise to vicinity of forward area airstrip
- Approach and Landing
- Park, unload, and reload
- Takeoff and climbout
- Cruise to secure base
- Land

For the purposes of this discussion, dispositions of conflicting surface forces can be described as one of the following:

- Fixed forward edge of battle area (fixed FEBA)
- Enclave

A fixed FEBA is roughly equivalent to a "front line," behind which the area is secure. An enclave is a secure area surrounded by an area open to enemy action.

The leakproof FEBA situation requires that threats be launched from the opposite side of this FEBA, which limits the weapon types due to the required range, and also limits the directions of approach.

The enclave situation places no minimum upon threat range. However, to continue to exist, an enclave must have control of the air and sufficient power to limit the enemy forces around the perimeter. Specifically, if the enemy perimeter forces are able to deliver explosive munitions onto the enclave in quantity, it will be lost. Accordingly, the enemy perimeter force may be defined to be of a guerrilla nature, using "shoot and scoot" tactics.

The minimum airfield usable by the medium STOL transport is on the order of 2,000 feet. Study of intratheater transport survivability resulted in definition of four classes of airfields, of which class III corresponds to the medium STOL minimum. A class III airfield is usually positioned 10 to 12 nautical miles from the FEBA. For their own protection, ground-based threats (artillery and rocket launchers) are usually positioned one-third of their range behind the FEBA, so that the class III airfield position requires a ground-based threat to have a minimum range of 15 to 18 nautical miles. A class III airfield in an enclave situation is predicted by the reference to be enclosed by a threat net similar to figure 126.

The threats to the STOL near a class III airfield as derived in the reference study are as follow:

1. Fixed FEBA

   a. Surface-to-surface threats (STOL on ground)

      - 130 mm gun, for which four to 12 rounds are predicted prior to reaction fire by friendly forces

   b. Air-to-surface threats

      - Single-place supersonic fighter, for which delivery of two to four 500-pound general-purpose bombs is predicted.

2. Enclave

   a. Surface-to-surface threats

      - 122 mm rocket, 18 to 54 rounds

      - 82 mm mortar, 15 to 40 rounds

   b. Surface-to-air threats

      - 14.5 mm, 12.7 mm, and 7.62 mm machine gun

      - 7.62 mm rifle

      - 23 mm and 37 mm weapons

A tactical SAM corresponding to the U.S. Redeye probably should be added to the list. This threat is an IR seeker with a contact fuzed small warhead and is usable by guerrilla forces.

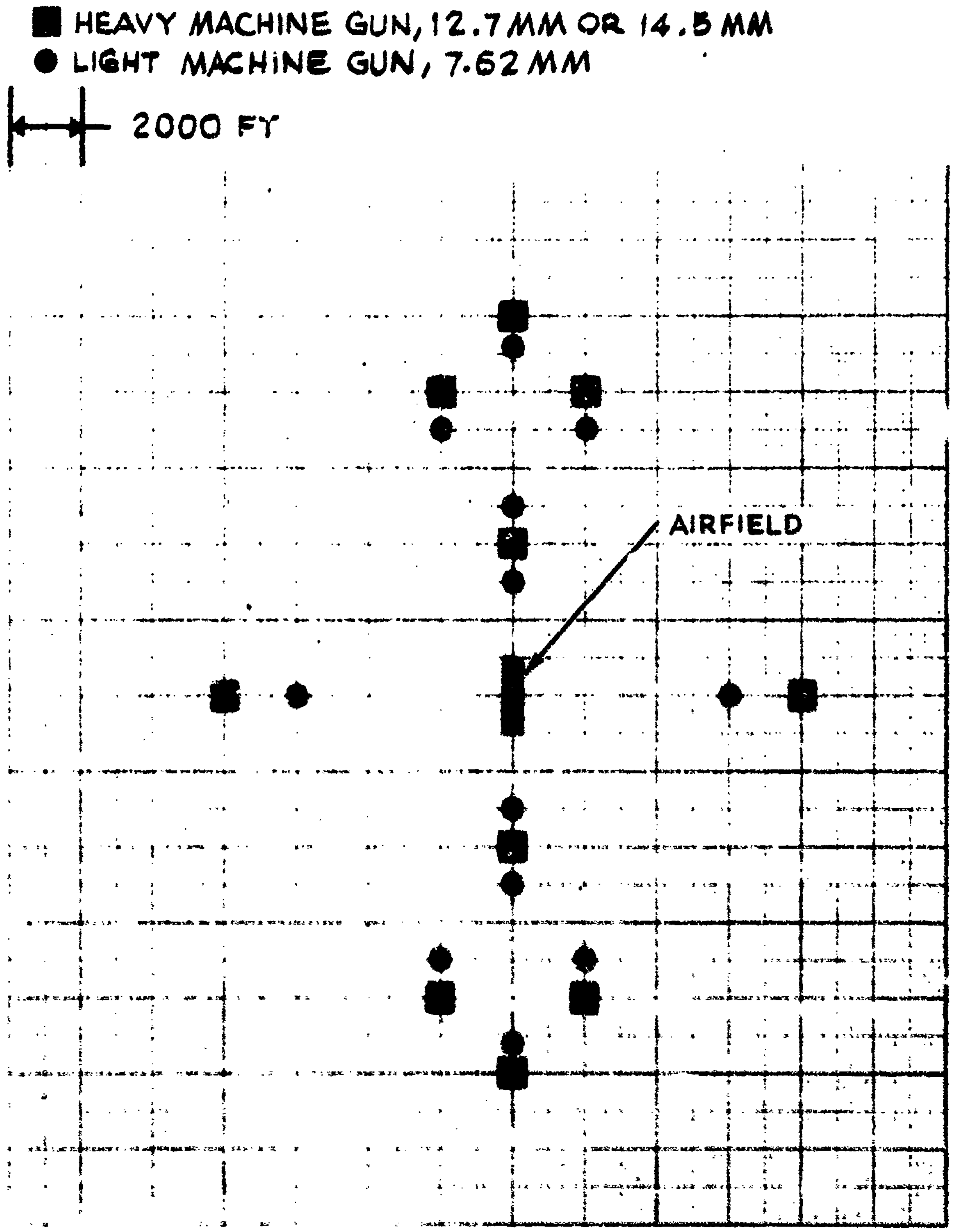


Figure 126. Representative Surface-to-Air Threat Net
Around a Class III Airfield

11.2 THREAT EFFECTS

The three most important kill mechanisms of the aforementioned threats involve ballistic blast overpressure, and incendiary effects. The combined blast and fragment effects resulting from the detonation of the larger mortar, projectile, or bomb threats are such that, at short burst-to-target distances, aircraft loss or damage requiring large times to repair are virtually certain. Consequently, the only feasible hardening techniques for these threats involve either shelters or dispersion of parked aircraft. The central objective of the survivability enhancement task for the aircraft on the ground can, therefore, be defined in terms of longer burst-to-target distances, at which blast pressures are reduced to a viable level. At such distances, penetration is the dominant effect.

11.2.1 PENETRATION

Comparison of the destructive potential of the fragments of explosive warheads and the inert rounds (bullets) of the expected threats shows that the 14.5 mm armor-piercing bullet respresents the maximum potential. This round is, therefore, selected to define the maximum penetrator protection requirement.

The expected threat to the transport while landing or taking off from the typical class III airfield is schematically shown in figure 127. Basically, these threats result in possible lower-hemisphere impacts on the transport which should be considered in placement of components and protection allocation to critical subsystems. The threat to the aircraft on the ground is primarily the multiple fragment penetration resulting from proximity detonations of mortar rounds, projectiles, and bombs. Because of effects such as ricochet, the fragment penetration could be expected from many varying angles. However, the protection features and design considerations for the in-flight cases are expected to result in significant protection from the partial fragmentation effect. The location and protection features such as self-sealing to prevent fuel from leaking on the ground and burning up an otherwise undamaged aircraft becomes vitally important for parked aircraft. Southeast Asia parked aircraft information supports this conclusion.

11.2.2 BLAST

For the threats considered, the blast overpressure on the structure is usually secondary to the multiple fragment penetration effects as far as lethal aircraft damage is concerned. Weight limitations prohibit protection of the structure from penetrators; hence, no blast requirement need be placed upon such structure. However, there may be areas such as cargo doors that are much more vulnerable to blast; these should be analyzed to insure that fragmenting warhead blast effects are secondary for all structure.

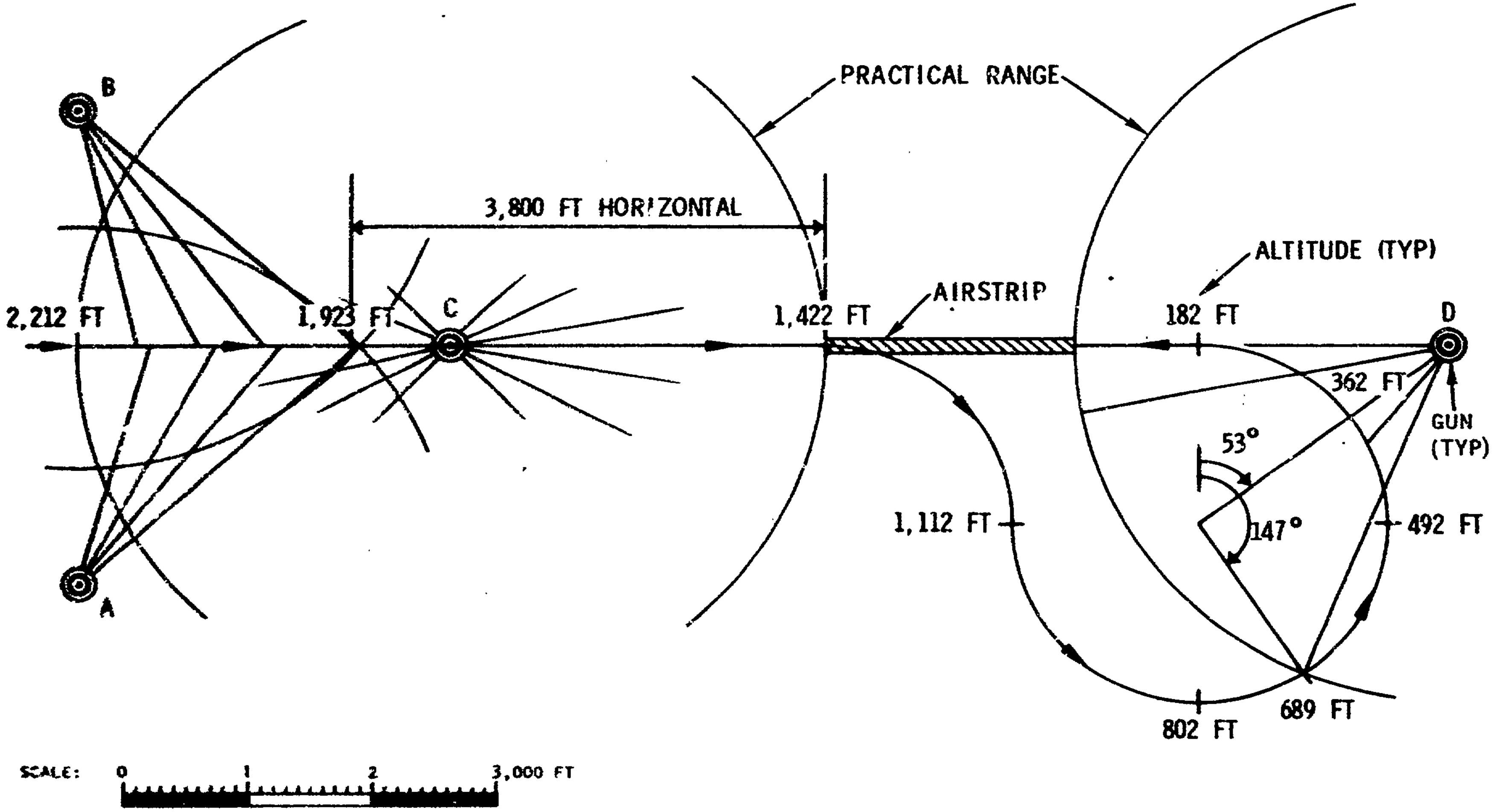


Figure 127. Heavy Machine Gun Threat to a STOL Landing at a Class III Airfield

The threat effect of a Redeye-type SAM is solely blast. This blast effect is so concentrated in area and of such magnitude that no sensible protection can be suggested. The hit probability from this threat can be lowered by reducing the STOL transport IR emission, particularly that of the engine exhaust.

11.2.3 INCENDIARY EFFECTS

A major portion of small-arms ammunition carries incendiary material that is activated by contact with a target. High-velocity fragments, impacting structure such as aluminum or titanium may generate their own incendiary flash (the so-called pyrophoric effects). Contact fuzed projectiles such as the 23 mm can also result in incendiary sources. The combustible materials of the STOL transport provide a damage mechanism by which a low-energy threat effect such as an incendiary bullet or high-speed projectile can disable or destroy the aircraft. Applicable protective measures can be taken to reduce the incendiary potential of an aircraft. The fuel and hydraulic subsystems, pyrotechnic device installations, crew compartment, engines, and cargo areas each require unique hardening techniques against incendiary mechanisms. Materials that require additional oxygen for burning, such as fuel, can be protected by reducing the probability that a fuel-oxygen (air) mixture will exist. Materials such as pyrotechnic devices and beathing oxygen must be shielded. The crew compartment can be protected by utilizing only materials that will not sustain combustion. The cargo area and the engines can be protected by fire extinguishing installations.

11.3 GENERAL REMARKS ON HARDENING

Threat effects that will destroy a STOL transport can be divided into two groups:

1. Catastrophic threat effects - those that inflict sufficient damage that the aircraft is lost

   Example: A 500-pound general-purpose bomb burst, 10 feet from the aircraft wingtip.

2. Triggering threat effects - those that damage a component in such a way that the component response results in aircraft loss.

   Example: A penetrator which causes a leak in the hydraulic system, causing loss of primary flight surface control and resulting in aircraft loss

Protection of the STOL transport against catastrophic effects is provided by airfield and air security measures. The main interest of hardening is toward the triggering threat effect. Triggering threat effects that apply to a parked STOL transport are those such as burning or detonation of the aircraft's fluid or pyrotechnics. A flying STOL transport is susceptible to these threats plus those that disable flight subsystems.

A major goal of the STOL transport hardening task is to minimize the probability at reasonable weight that a single penetrator can trigger aircraft loss. Since the penetrator threat is predominantly from the lower hemisphere, it follows that the more critical of the vulnerable components should be located in the upper portion of the fuselage, above the cargo space. Of these, the more critical should be located closer to the centerline.

Subsystem redundancies markedly enhance penetrator survivability; however, for this to be effective, the redundant components must be separated so that no penetrator could disable more than one of them.

11.4 SUBSYSTEM HARDENING

11.4.1 STRUCTURE, INCLUDING CONTROL SURFACES AND LANDING GEAR

One structural design objective is that it be designed to accept one penetrator (14.5 mm armor-piercing at 2,400 feet per second) without degradation of any essential capability. Another objective is that the structure be designed to perform a part of the shielding function. For an example of this last, the lines and controls in the vertical stabilizer and engine pylons are concentrated into small cross sections and are close to the mold line. In these locations, lines that are parts of redundant hydraulic systems are so confined that one penetrator could cause multiple system disablement. Box section structure can be designed to provide a considerable measure of protection. Generally, the structure should be designed with redundant load paths and using shatter-resistant materials. These will reduce the probability of aircraft structural damage resulting in aircraft loss. An additional important factor to be considered is battle damage repairability, since an aircraft requiring extensive man-hours to repair is essentially a "lost" aircraft.

11.4.2 CREW AND CREW STATION

The crew station is shielded from direct rear angles by the fuselage and its contents, and to some extent from below. This shielding can be augmented by removable armor that will protect from all angles of the lower hemisphere and on the order of 15 degrees above the horizontal. Protection from forward directions is the most serious problem; here, the requirement for armor protection is opposed by the requirement for the pilots to use the controls, read

the instruments, and observe through the windows. For this reason, it is preferable to install armor forward of the instrument panel and instruments. Provisions for armor installation and removal must be developed early in the design.

The location of the forward armor makes a box around and under the crew more practical than individual armored seats or crewman enclosures, as designed for the C-130. The box will also protect the flight-essential components in the crew station, as a crewman enclosure would not.

Armor protection in way of the windshield can be provided by external bolt-on armor, windshields made of transparent armor, or internal bolt-on. The bolt-on armor could be transparent, or opaque with transparent insets. One aspect of this problem is that transparent armor that is effective against a 14.5 mm penetrator will be both bulky and heavy. Transparent inset might be designed to lesser requirements if the areas required were sufficiently small.

11.4.3 FUEL SYSTEM

Hardening of the fuel system involves consideration of protection from explosion in vapor-filled tanks, leakage and fire, catastrophic fuel leakage, and fuel loss such that engine fuel starvation is experience. In an unprotected fuel system, a very high probability of achieving one of these failure mechanisms exists given a hit by any threat. Since the presented area of the fuel system is very large from most aspects of interest, this presents a large vulnerability problem. The fuel supply systems should be laid out so that they are buried in the tanks as much as possible and that no one hit on a tank or line will result in loss of engine feed capability. All point designs should include these design considerations as well as fire, explosion, and leakage suppression. Additionally, careful attention will be given in the design to fasteners and tank attach points, with the net result of all these considerations being a crash-resistant as well as a vulnerability-resistant design.

11.4.4 PROPULSION SYSTEM

The dominant survivability feature of the STOL transport propulsion system is the redundancy provided by the four engines. The fire extinguishing system is a second important feature. Since the STOL transport is not a combat aircraft, the redundancy obviates the need for protection features that add more than a slight weight increment. However, the following features can be considered:

- Provide high-velocity ventilation and high-rate fluid drainage of potential fire areas

- Provide location of fluid lines and other components having a high fire potential toward the top center of the pod or in some other more protected location

- Provide for a two-shot engine fire extinguishing system

11.4.5 HYDRAULIC SYSTEM

Hydraulic fluid in a large line or tank can be exploded or ignited by an incendiary penetrator. The use of redundant systems minimizes the effect of fluid leaks, but does not protect against combustion. Accordingly, fluid lines and components should be accorded high priority for protection against penetration. Electrical power can be considered for applications wherein hydraulic and electric power are competitive. The hydraulic system in each of the point designs includes four completely independent systems with two pumps per system. Maximum system separation is used to minimize the probability that one hit will cause failure of more than one system.

11.4.6 FLIGHT CONTROL SYSTEM

The redundancies that are built into the system are made useful for survivability by separation of the members of redundant sets. The concept used for the basepoint flight control is that the system will provide 100-percent capability following one failure, and safe operation with reduced capability following two failures. Primary flight controls are powered by four completely independent hydraulic systems, each system powered by two hydraulic pumps, and any three systems able to provide maximum control system capability.

11.4.7 ELECTRICAL SYSTEM

The two general methods of electrical power survivability enhancement are separation of redundant components and protection of all components by placement behind heavy structure and less critical components. The circuit protection, built into the system for other reasons, provides a considerable protection against short and open circuits caused by penetrators, fire, etc.

The system selected for the point design insures that deactivation and/or shorting of any one component by threat effects will not trigger a large-scale system failure. Areas such as phase synchronization and essential bus contactors have been designed with consideration for this condition.

The effects of penetrating storage batteries require that these receive careful attention, with respect to the circuit effect and the corrosive fluid that may leak through the hole in the compartment made by the penetrator.

Four interconnecting systems have been used in the basepoint design with a generator on each engine. Full system capability is retained with loss of any two generators. Bus separation and redundancy provide power distribution to all electrical components from any two generators.

11.4.8 BATTLE DAMAGE REPAIR

The foregoing design considerations will result in an aircraft that is capable of surviving hits from many threats. Another important consideration in the design is to allow rapid repair or replacement of nonlethal battle damage. These considerations are just as important as reductions in physical vulnerability since an aircraft requiring several thousands man-hours to repair is just as effectively "lost" to the commander as one that has received lethal damage.

SECTION XII

TRADE STUDIES

The STOL TAI design requirements, as outlined in section III, specify several design trades for which vehicle effects are to be determined. These trades are as follows:

Trade	Basepoint Design	Trade Value
Mission radius (n mi)	500	750
Cruise speed (M)	0.75	0.85
Field length (ft)	2,000	±500
Cargo bay size (ft)	12 x 12 x 45	12 x 12 x 55
Sea-level penetration (knots)	-	400

In addition to the foregoing trades, a weight growth trade was conducted to determine the effects dead weight has on takeoff gross weight (TOGW).

The trade studies discussed in this section are applicable to the vehicle (D-516-2A) described in section IV. The trade studies originate about the analytical model of the D-516-2A baseline configuration determined by the results of the sizing exercise. Each trade was conducted by incorporating the appropriate change in weight, geometry, or aerodynamics on the sizing program basepoint and the resizing to the desired performance level depending on the trade values of interest. Sizing thrust-to-weight ratios and wing loading combinations for the various design condition are depicted in figure 39, which indicates the various design conditions and the vehicle conditions that solve a particular trade value.

12.1 WEIGHT GROWTH TRADE

The purpose of the trade is to establish a growth factor for the dead weight effects on vehicle TOGW. This trade involves arbitrarily incrementing the fixed load weight and resizing the vehicle to regain performance. The results of the trade indicate that vehicle sensitivity to dead weight is approximately 1.95 pounds of TOGW for each pound of dead weight. The 1.95 growth factor is somewhat lower than that of the preliminary baseline value of 2.7, but is to be expected due to the change in design criteria governed by

the different takeoff and landing ground rules. The preliminary baseline configurations (part 1) dominant sizing conditions were the takeoff distance of 2,000 feet over a 50-foot obstacle with a 3-degree gradient, and the wing sized by the fuel requirement imposed by the 2,600-nautical-mile range mission. The takeoff sizing produced higher thrust-to-weight levels and, therefore, higher growth through larger engine sizes. The refined baseline is sized by the same 2,600-nautical-mile fuel requirements and the 0.75M speed requirement at 20,000-foot altitude. Another significant change since the preliminary baseline is a change in engine from the 7.8 bypass ratio (BPR) GE13/F4B to the 6.5 BPR GE13/F10 engine. The GE13/F10 engine is an advanced technology engine and has an improved thrust-to-weight ratio which would tend to reduce the growth factor. The results of the weight growth trade are shown in figure 128.

12.2 FIELD LENGTH TRADE

The field length trade influences the aircraft primarily through the wing loading required to reduce the landing distance. The refined configuration landing distance over a 50-foot obstacle is 1,675 feet with balanced field takeoff distance of 1,250 feet. The fact that the basic vehicle is being sized for conditions other than field length means that all distances in excess of 1,675 feet are satisfied by the same vehicle. Vehicle gross weight changes for distances above 1,675 feet can only be expected by relieving the 0.75M speed requirement at 20,000-foot altitude. If this requirement is waived and the takeoff balanced field length is the sizing variable, then vehicle gross weight reduces approximately 8,000 pounds. This, however, would cause a reduced cruise speed and involves changes in design requirements. The trade study shown in figure 129 holds all the design criteria except the field length requirements. As shown, there is no change in TOGW until the required field length is below 1,675 feet. At a 1,600-foot field length, TOGW would increase by approximately 4,000 pounds, and at 1,500 feet, a 12,000-pound increase may be expected. Reduced field length may be obtained by increasing thrust to weight in order to reduce velocity at the obstacle, but this method produces vehicle weight increments five to six times as large as by wing loading reductions. The vehicles sized to the reduced field lengths by wing loading reduction do have an excess in wing fuel volume requirements imposed by the 2,600-nautical-mile range mission. The additional wing fuel volume would improve the range available on the zero cargo range mission which would improve unrefueled deployment capability.

12.3 DESIGN MISSION RADIUS TRADE

The design mission radius trade involves an increase in the design mission radius from 500 to 750 nautical miles and still maintain a rate of

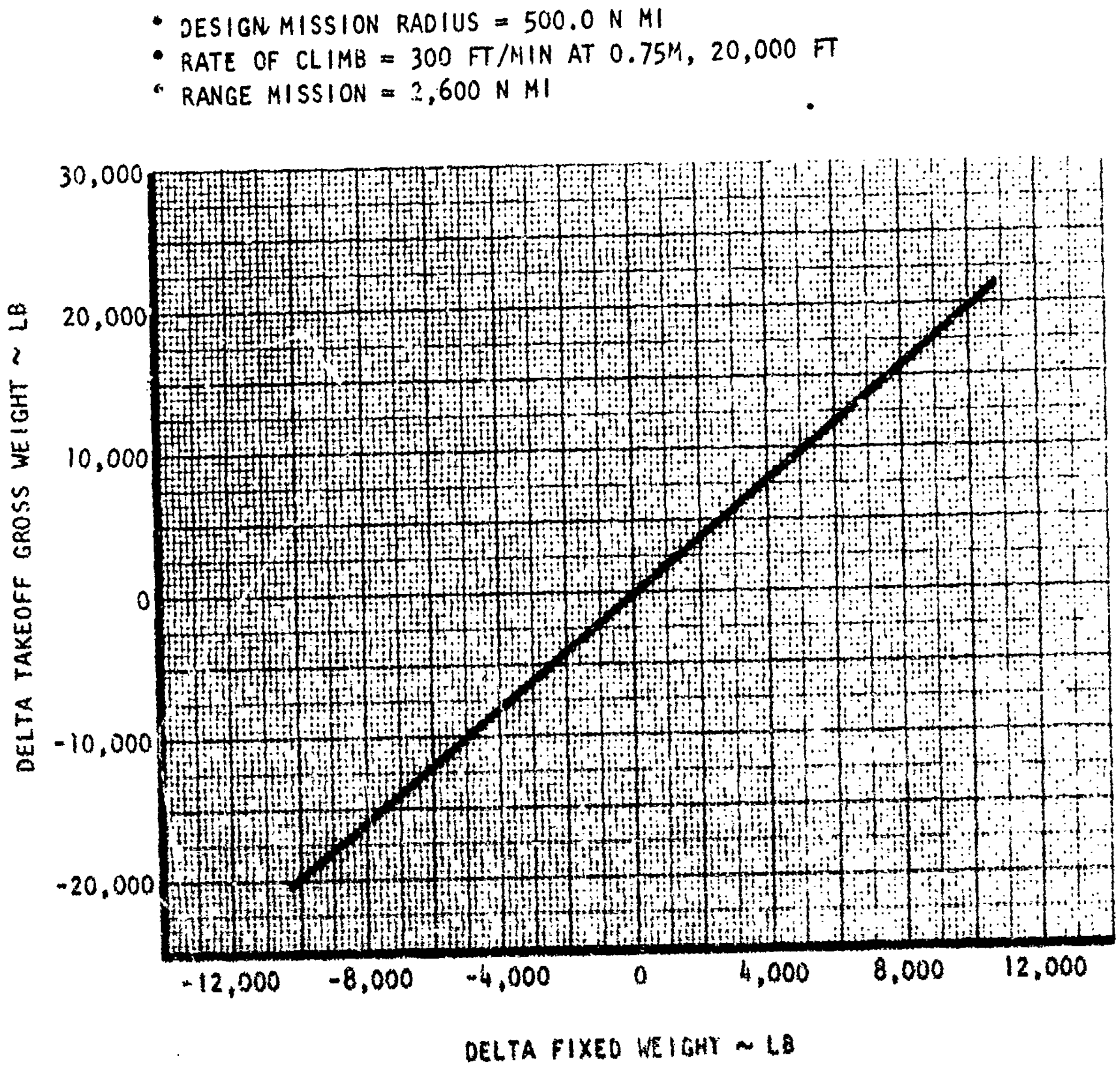


Figure 128. Weight Growth Trade

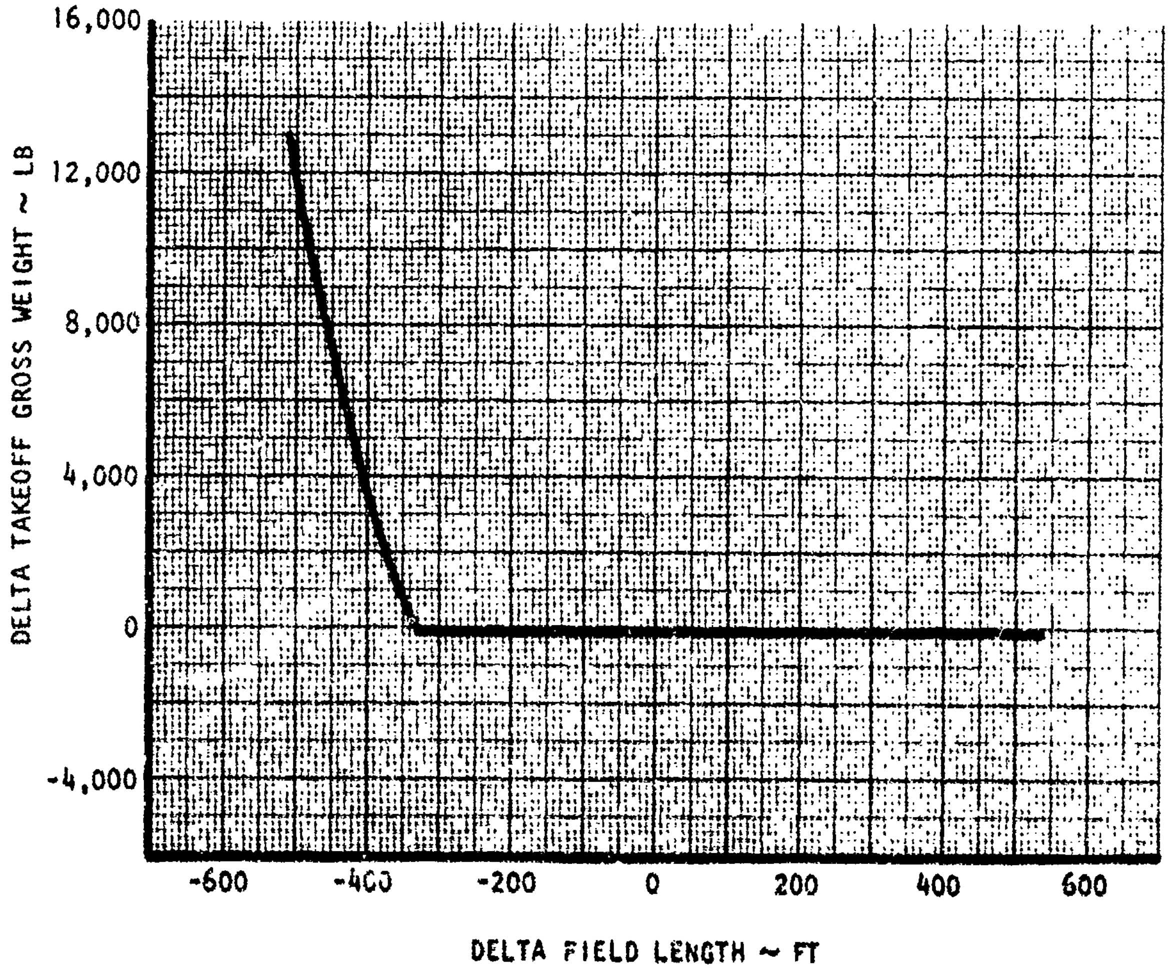


Figure 129. Field Length Trade

climb of 300 feet per minute and 0.75M cruise speed at 20,000-foot altitude. The takeoff and landing distances at the resized vehicle midmission thrust-to-weight ratio of 0.53 and wing loading of 91.85 pounds per square foot are approximately 1,300 feet and 1,700 feet, respectively. The midmission weight was based on TOGW minus approximately 44.8 percent of the fuel required. The result of this trade is shown in figure 130, indicating a resized vehicle with a TOGW increment of 19,197 pounds to meet the 750-nautical-mile design radius. The resized vehicle with the increased radius performance results in a 13.85-percent fuel fraction (46,476 pounds of fuel) and a 1,713-square-foot wing area.

12.4 CARGO BAY LENGTH TRADE

The cargo bay length trade (45 feet to 55 feet) was evaluated by increasing fuselage length with an addition of a constant-diameter 10-foot-long section and relocating the wings and tails. It was assumed that the other aircraft components could be rearranged such that the major effect would be directly associated with the fuselage geometry. The additional fuselage length results in a 11.3-percent increase in wetted area and a 3,180-pound vehicle weight increment. The basic fuselage weight increased by 2,350 pounds along with an 830-pound increase in furnished weight. The furnishing weight increase is for the additional cargo handling equipment and troop-carrying capability associated with the longer fuselage. The previous geometric and weight change, along with the increased fuselage friction drag, were incorporated in the sizing program baseline configuration. The vehicle was resized to meet the 500-nautical-mile design radius mission, the 0.75M cruise requirement at 20,000 feet and fuel available - required wing volume condition for the 2,600-nautical-mile range mission. The resized vehicle with the increased cargo bay size results in an approximate 8,270-pound increase in vehicle TOGW. Trade results are shown in figure 131. This trade, when conducted on the part I preliminary baseline configuration, resulted in a 7,500-pound gross weight increment. The difference in weight increments results from study ground rule changes, and incorporation of the structural design studies conducted during part II study phase.

12.5 CRUISE SPEED TRADE

The results of the cruise speed trade are presented in figure 132. This trade involves changes in wing characteristics to improve drag divergence for the increased cruise speed. The wing characteristics that are required are a 10-percent thickness ratio across the entire span and quarter-chord sweep increase from 25 to 39 degrees. Aspect ratio was reduced to 7.0 from 8.0 to

offset increased wing weights caused by increased sweep and reduced thickness. Other vehicle changes would not be required because of the fuselage and empennage drag divergence being higher than 0.85 mach number. The wing changes result in a 4,001-pound increase in the basic wing weight prior to resizing. The predominant sizing conditions for this trade were the 500-nautical-mile design radius mission, the 0.85M speed requirement at 20,000-foot altitude and fuel available - required wing fuel volume constraint of the 2,600-nautical-mile range mission. Trade results indicate that the 0.1 delta mach number cruise capability would add approximately 51,000 pounds to TOGW. Pertinent vehicle characteristics are initial thrust-to-weight of 0.663, wing loading of 104.5 pounds per square foot. The resized vehicle has four 157.7-percent size GE13/F10 engines and a 2010.0-square-foot wing area. Midmission thrust-to-weight is 0.707 with a 94.5-pounds-per-square-foot wing loading.

12.6 SEA LEVEL PENETRATION SPEED

The baseline vehicle design q was established by 0.75M cruise speed capability at 20,000 feet and was the governing factor for a maximum cruise speed below 20,000 feet. The baseline vehicle resulted in a sea-level cruise speed of 311 knots. Increasing the sea level cruise speed to 400 knots, the fixed-weight penalty of the baseline vehicle (weight increment times the growth factor of 1.95) would result in a TOGW increment of 510 pounds to maintain a constant baseline performance.

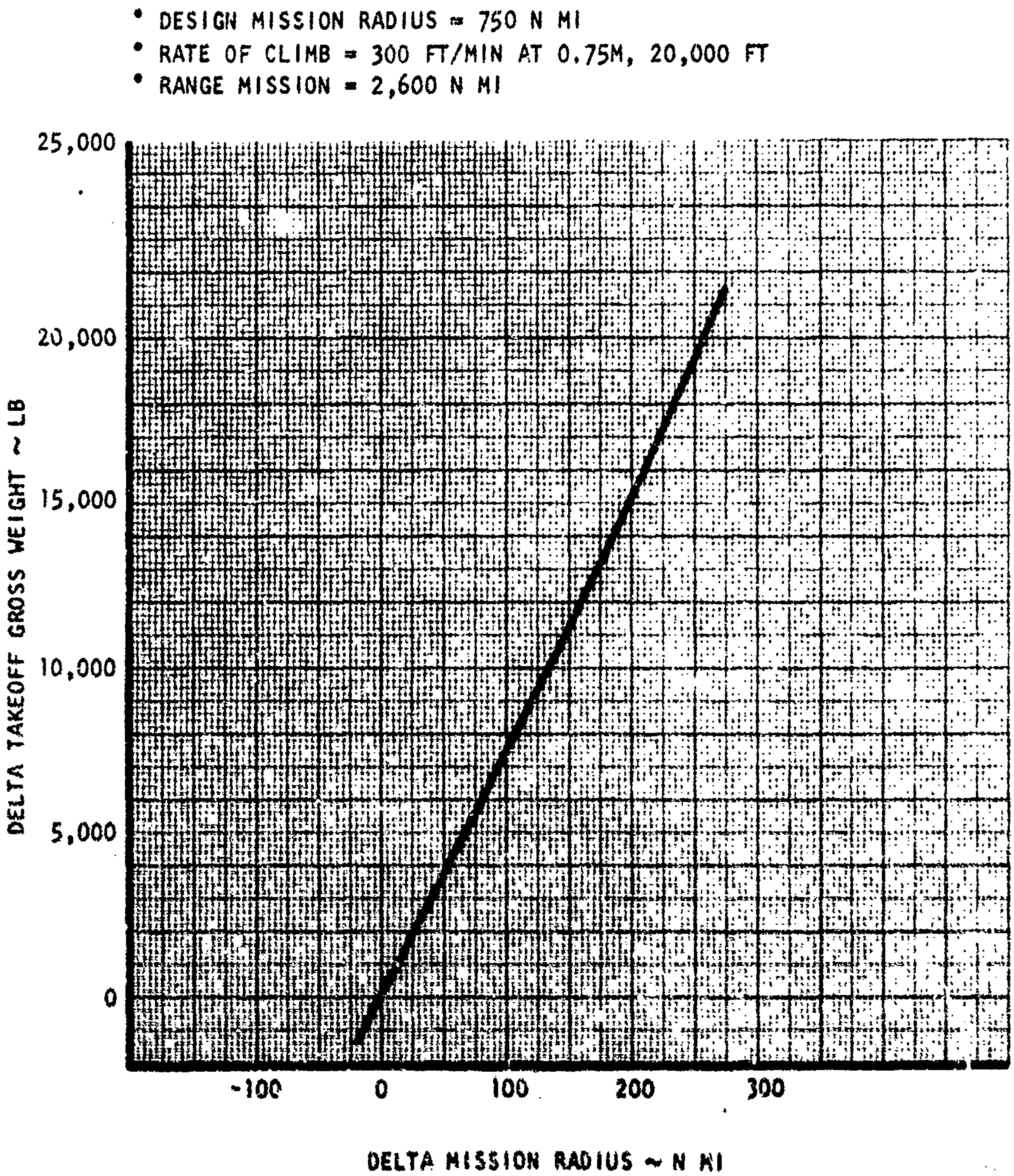


Figure 130. Design Mission Radius Trade

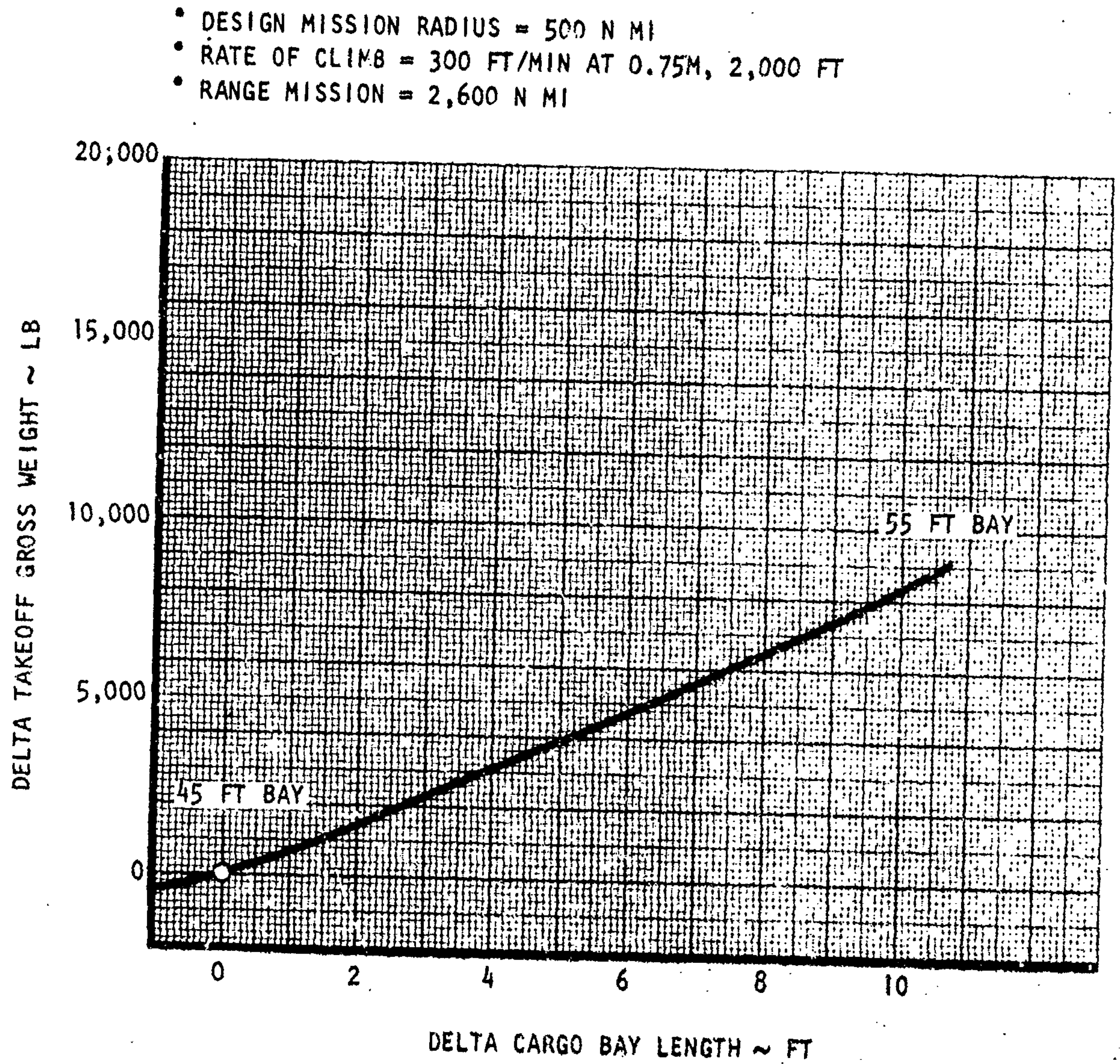


Figure 131. Cargo Bay Length Trade

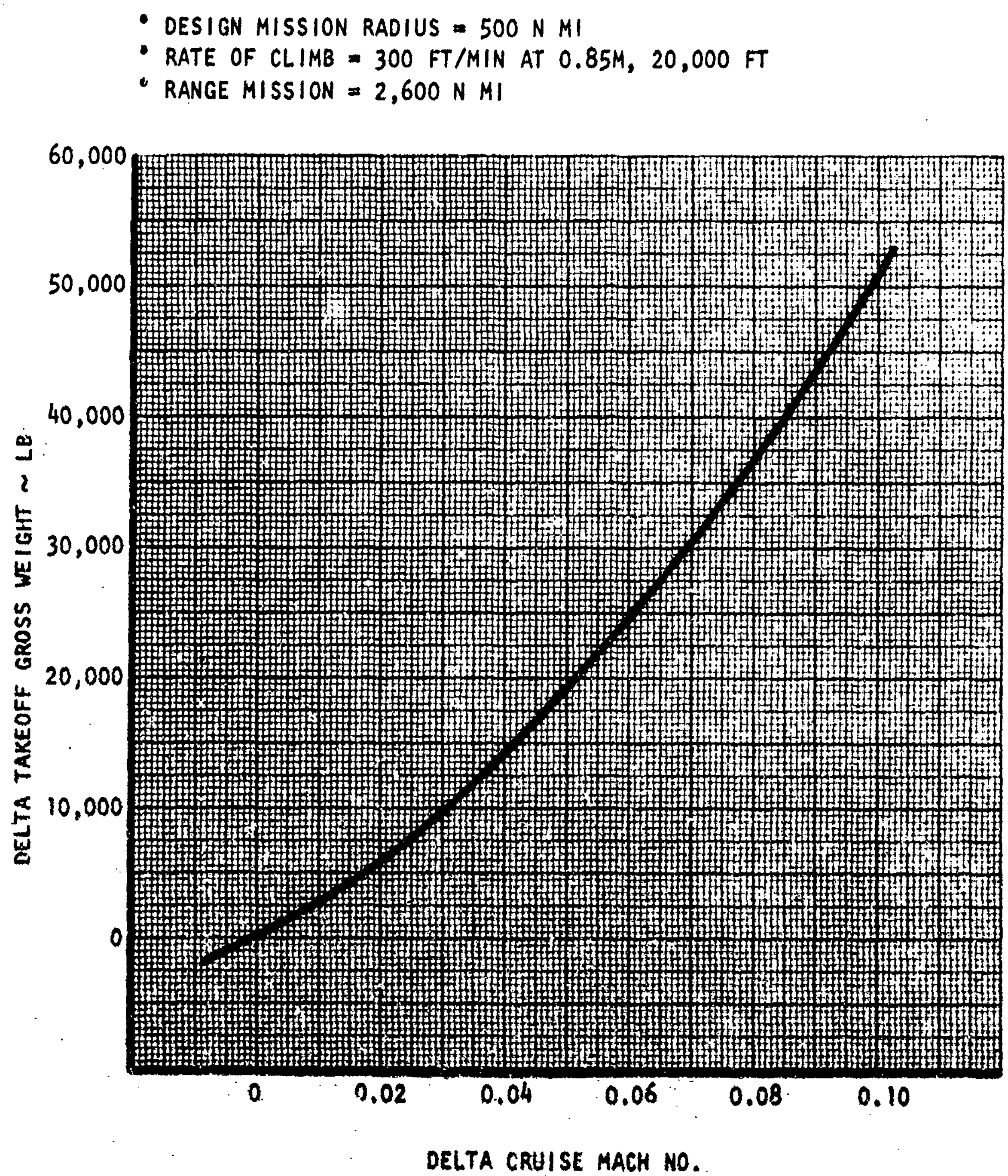


Figure 132. Cruise Speed Trade

Section XIII

ADDITIONAL TECHNOLOGY REQUIREMENTS

The primary objectives of a technology program supporting the development of the MST are (1) to provide an accurate basis on which the decision for final airplane requirements are made, (2) to provide the technical capability to meet these airplane design requirements on a low-risk basis, and (3) to advance the state-of-the-art in the STOL-related technologies so that the range of selection of airplane requirements can be broadened. To assure a successful production model of the MST, it is necessary to pursue these objectives through the formulative phases of the program until firm requirements and a final airplane design are established.

The present technology program has developed aerodynamic design techniques and conceptual configuration designs to the degree that current performance requirements can be met with a high confidence level; however, this present work has not been directed at configuration optimization, and it is believed that performance improvements can be achieved. Other technical areas, specifically flight control techniques and structural design criteria, will require additional effort to assure a low-risk approach to a production MST program. All of the STOL technology areas should be continued in order to be responsive to changing requirements, and to develop improvements which will be reflected in a more effective product in the eventual MST program.

Fulfillment of the basic objectives of the STOL technology investigation program will require continuing action in the following areas:

1. Configuration analyses to provide up-to-date trade effects of variations in requirements and alternate design approaches.

2. Engineering analyses, supported by wind tunnel, flight simulator, and laboratory tests to refine the capability of translating varying airplane requirements into configuration and subsystem design details.

3. Engineering studies directed at achieving higher efficiencies in performance, flight procedures, structural design, and cost.

Specific tasks which should be pursued on a near-term basis have been identified in the categories of:

1. Configuration and subsystem design

2. Aerodynamic characteristics

3. Structures and mass properties

4. Flight control systems

13.1 CONFIGURATION AND SUBSYSTEM DESIGN

13.1.1 CONFIGURATION TRADE STUDIES

Continuing configuration studies are required to determine the quantitative effects of changes in requirements, or suggested alternate design approaches. A configuration design should be maintained up to date to meet all current performance requirements. Configuration changes required to comply with requirements changes, and investigations of new concepts, should be applied to the baseline configuration and the impact on airplane weight and performance determined.

13.1.2 BLC AIR SUPPLY

More stringent STOL performance requirements may necessitate the use of BLC on the flight control surfaces of the MST to provide satisfactory handling qualities under adverse conditions. Engine bleed air is normally used as a source of high-pressure air for such purposes; however, large bleed-air demands from high bypass engines result in large engine performance penalties. It is therefore recommended that an alternate air source be evaluated in terms of overall impact on the total airplane. Factors to be considered include weight, reliability, space requirements, growth potential, and utility.

13.1.3 GROUND MOBILITY SYSTEM

The desired ground mobility capability should be evaluated to determine whether a STOL-type transport has sufficiently improved operational capability with this system to justify its additional cost and modified aircraft flight efficiency. To attain these answers, it is recommended that a three-phase program be initiated that would include an alternate design concept trade study, detail design and construction of the selected system, and an actual flight demonstration program.

The phase I effort would be directed toward a general broad scope study to show the trade-off factors of weight, performance, and reliability for all design approaches. Phase II would be the detail design of the most promising system as applied to a demonstration aircraft. The aircraft should be a military transport in the 130,000- to 190,000-pound weight class and having a four-wheel bogie-type main gear. If necessary, the aircraft may be restricted to flying with gear extended at all times. Phase III should consist of actual construction of the demonstration system, installation on the test aircraft, and operational tests of the system.

13.1.4 ENGINE NOZZLE DEFLECTOR

Current investigations show large performance benefits of a controllable engine nozzle deflector that operates between approximately ±15° deflection. For application to an MST, this nozzle must have a reasonably high actuation rate, minimal thrust losses, and high reliability. A study should be initiated to evaluate deflection nozzle concepts and actuator systems to fulfill these requirements at minimum weight consistent with the necessary reliability.

13.2 AERODYNAMIC CHARACTERISTICS

13.2.1 AERODYNAMIC CONTROL OPTIMIZATION

The present study has provided adequate data for the design of pitch, roll, and yaw controls to meet the current airplane requirements. However, a critical safety margin based on the stall speed with a failed engine, in the presence of the ground, requires further work on asymmetric stall control by leading edge devices, and ground effect on $C_{L_{max}}$.

Present investigations showed lower roll control effectiveness using spoilers with triple-slotted flaps than with double-slotted flaps. This characteristic penalizes the use of triple-slotted flaps which may be desirable from other considerations. Further investigation in this area might yield a reasonably simple solution to an improved flap and roll control system.

The general solution to an adverse downwash problem associated with very high-lift coefficients has been to move the horizontal tail up and forward, requiring a larger tail area and resulting in a structural weight penalty. Some recent NASA data show some promise of use of a lower tail location. More extensive investigations of the effects of configuration geometry (wing and tail aspect ratio, engine location, tail location, sweep) are required to be able to optimize the tail geometry and location.

13.2.2 AERODYNAMIC DESIGN METHODS

The lift developed by blown flaps depends directly on the jet turning angle and jet turning efficiency. Values of these parameters are determined by static tests or are estimated by methods described elsewhere in this report. These methods yield only the results of tests of exploratory wind tunnel models where no optimization was attempted. Studies should be made to develop a better understanding of the factors which affect the jet turning process, so that a systematic approach could be taken in optimization of this process.

13.2.3 PERFORMANCE OPTIMIZATION

The present program also investigated the effects of nacelle location and engine orientation on lift performance. These results indicate better performance with the nacelle closer to the lower surface of the wing which is, in general, contrary to good, high-speed performance. Means of obtaining the best compromise for low- and high-speed performance should be investigated. Although the effects of engine orientation were determined for three thrust axis angles in the present study, additional tests are required to determine an optimum. Thrust inclination angles of -12, 3, and 18 degrees were tested, with the 18-degree angle giving the best performance. It is believed that further improvement in performance can be obtained by testing perturbations around the 18-degree setting.

To date, the propulsion-lift systems have been tailored toward improved low-speed performance. Consideration of cruise speed performance has imposed some constraints on configuration synthesis, but this has been on a "judgement" basis, rather than on a quantitative test basis. High-speed, as well as low-speed, tests should be made to determine the effects on overall mission capability of nacelle arrangement and other STOL transport features such as gear pods and aft fuselage shaping.

13.3 STRUCTURES AND MASS PROPERTIES

Detailed structural design criteria for components of powered lift systems have not been developed at the present time. These structural components are subjected to a nonconventional environment and do require specific design criteria.

Also dependent on these structural design criteria is the development of more accurate weight estimation techniques for STOL aircraft. These procedures are based on the structural design techniques applied to meet the design criteria.

A program is recommended to develop preliminary design techniques from a survey of available data relative to design criteria applicable to STOL blown flap structures and then to obtain test data to validate and refine the criteria developed. Elements of the recommended program are described in the following paragraphs.

13.3.1 DESIGN CRITERIA

Optimum and complete design criteria are required to properly design efficient blown flap structure. New and likely higher levels of acoustic fatigue criteria and dynamic buffetting loads are caused by the interaction of engine exhaust impingement and the normal oscillating aerodynamic forces associated with high-lift flap systems. The potential significance of material erosion or accelerated oxidation, etc, are not known. Because of the near-term need and potential significance to the development of efficient and flightworthy flap structures, this task should receive priority consideration, even though the system application is limited.

13.3.1.1 Recommended Approach

The recommended approach is as follows:

1. Assemble a monograph from available published information on noise prediction, aerodynamic buffetting, applicable material properties, data, and techniques relevant to blown flap design problem. The monograph should include data on all applicable significant parameters and phenomena. For example, data should be assembled on boundary layer noise generation mechanisms, edge tones, effects of shed vortices, slots, nozzle geometry, erosion and oxidation characteristics of candidate materials, etc.

2. Synthesize typical promising blown flap configurations in terms of propulsion and flap design and operating parameters.

3. Apply data and techniques of step 1 to provide preliminary definition of noise, buffetting, and exhaust impingement environment for cases represented by step 2.

4. Investigate amelioration techniques (e.g., velocity suppression, sound dispersion, shielding, noise frequency shifting) and identify resulting new noise/buffet/exhaust impingement characteristics and design criteria.

5. Based on steps 1 through 4, define preliminary flap system acoustic fatigue, buffet loading spectra, and exhaust impingement design criteria for parametric conditions embracing environments of both steps 3 and 4. The criteria should embrace the full range of potential environmental conditions possible through design options, with respect to material temperature, structural concept, exhaust velocity, and jet exhaust/flap geometry, etc.

6. Perform preliminary design and cost studies of typical flap and flap retraction/retention systems, based on representative rational designer options within range of design data provided by activity through step 5. These studies will translate the design criteria resulting from each selection of design approach into a definition of the resulting effect on the structural weight and cost.

7. It is not expected that steps 1 through 6 can be accomplished with uniformly high confidence levels, because of the current status of data and applicable techniques. Appraise relative success achieved at each earlier step, identify deficiencies, and recommend further analytical and test programs to rectify critical deficiencies or to direct research into promising high payoff technology development areas.

13.3.2 INVESTIGATION AND TEST OF HIGH-LIFT SYSTEM COMPONENTS FOR STOL AIRCRAFT

This program is planned in three phases as follows:

1. Perform structural concept and cost evaluation studies of high-lift systems (flaps, slats, and spoilers) to determine most cost-effective structure for these components. Included should be the mechanical systems such as the tracks for flaps and slats, since these represent as much weight and cost as the components themselves. Selective reinforcement, using advanced composites should be considered, particularly as a means of stiffening flap and slat tracks.

2. Perform element and structural tests of proposed concepts to prove feasibility of proposed construction methods.

3. Construct working test model of high-lift system, using construction concepts developed from steps 1 and 2 tasks. The test model should include flaps, slats, simulated wing box, and engine. Test will prove the ability of proposed concepts to withstand environment, including engine noise and pressures, temperature, structural loads, and stiffness requirements under actual operating conditions.

13.3.3 INVESTIGATION OF ADVANCED COMPOSITES APPLICATIONS TO STOL AIRCRAFT

This study will determine the cost-effectiveness of advanced composites on STOL aircraft structures. Although some of the previous studies may include the use of selective reinforcement with composites, this study will include the use of composites as the primary load-carrying members. They should be considered with composite substructure, or in combination with metallic substructure. Their effectiveness should be compared with conventional construction methods, as well as with the advanced construction methods of the previous tasks. This study should include primary structure such as the wing and empennage boxes and fuselage shell, as well as the flaps, slats, rudders, and elevators. Both boron and graphite composites should be considered.

This study should be coordinated with the foregoing and completed before the construction of test hardware, since composites may prove cost-effective in some of these structures. This would affect the test programs of the previous items.

13.3.4 WEIGHT ESTIMATION

The STOL vehicle investigation revealed some problems in the area of weight-estimating technology. These problems are basically unique to STOL application — blown flap load effects on torque box structure, blown flap components, double-hinged control surfaces, etc. The weight impact of these items cannot be accurately predicted with existing weight-estimation methods. Investigations in the following areas should be accomplished to improve the confidence level of the estimated weights:

1. Effects of blown flaps criteria on flap and torque box weights

2. STOL propulsion acoustic fatigue requirements on structures

3. Landing load spectrum on fuselage and gear structures

4. Flight control actuation system for double-hinged surfaces

5. Thrust deflector nozzle weights

13.4 FLIGHT CONTROL TECHNIQUES

13.4.1 FLIGHT SIMULATOR TESTS

The factor limiting STOL performance at the present time is the ability to safely control the flightpath with large upsetting moments associated with STOL operations at the low dynamic pressure. A further contribution to the severity of the problem is the desire to operate out of a 60-foot-wide landing strip. Initial studies in the present program brought out the extreme difficulty encountered in landing on a 60-foot-wide strip with strong crosswinds. A systematic approach to investigating the overall influence of landing strip width and the further development of piloting techniques should be made.

The following study areas are recommended for future flight simulation studies and are based on the present (1972) state-of-the-art in STOL technology. The study areas recommended have been selected because of their applicability to generalized STOL configurations. In all cases, the results will provide usable design guides in terms of piloted handling qualities.

For preliminary evaluation of these areas and determination of generalized design guides, flight simulation studies are preferred to flight test results. The direct control of independent variables which can be achieved by simulation assures design guides that are independent of all other variables, except the one being investigated. Subsequent correlation of simulation results with flight test data, where available, is, of course, necessary and desirable to assure accuracy of the study results.

1. Definition of a method for establishing pilot control techniques and evaluation of the effect of variations in technique on STOL landing and takeoff mode handling qualities. Both normal and failure mode conditions should be examined, as well as operation near the maximum maneuver limits of the aircraft.

2. Definition of the influence of allowable failure forces and moments on STOL handling qualities in terms of pilot ratings.

3. Evaluation of the influence of simultaneously required maximum control powers and/or maximum rates in all axes.

4. Evaluation of the effect of flap rate on pilot controllability during transition from conventional to STOL mode operation and during waveoff.

5. Definition and evaluation of STOL IFR display requirements.

6. Evaluation of the effect of IFR conditions on aircraft safety margin requirements.

7. Definition of ground effects on pilot handling qualities and control requirements for normal and failure mode operation.

8. Evaluation of the effects of landing strip width on control requirements and pilot technique.

13.4.2 ADVANCED CONTROL SYSTEM CONCEPTS

The possibility of obtaining benefits in flying qualities, performance, and total system costs, through the use of advanced technology in flight control systems, should be investigated.

Specifically, the impact of applying the control configured vehicle (CCV) approach to the MST should be evaluated. Also of interest is an automated engine failure compensation system which would employ control surfaces, engine modulation, and nozzle deflectors to control the large moments due to engine thrust loss.

13.5 FLIGHT TESTING

Many aspects of operation and design refinement of a STOL transport system can be investigated only (or most effectively) by actual flight testing of a full-scale airplane. The initial planning for the MST/ADP included a flight demonstrator airplane program to provide this capability. The test airplane was to be a full-size representation of the MST with nearly full capability, but using engines which would be available earlier than the planned MST engine. It was a requirement of this program to provide a conceptual design of a flight demonstrator airplane, a test program, and budgetary costs and schedules. Since the present STOL/TAI program was initiated, an AMST prototype program has evolved which preempts many of the objectives of the flight demonstrator. Because of this, requirements for a flight demonstrator plan were deleted. However, since a good portion of effort on the flight demonstrator had been accomplished by the time this

decision was reached, and since the scope of the prototype program has not yet been fully determined, a nominal program for a flight demonstrator is presented for possible use in future planning.

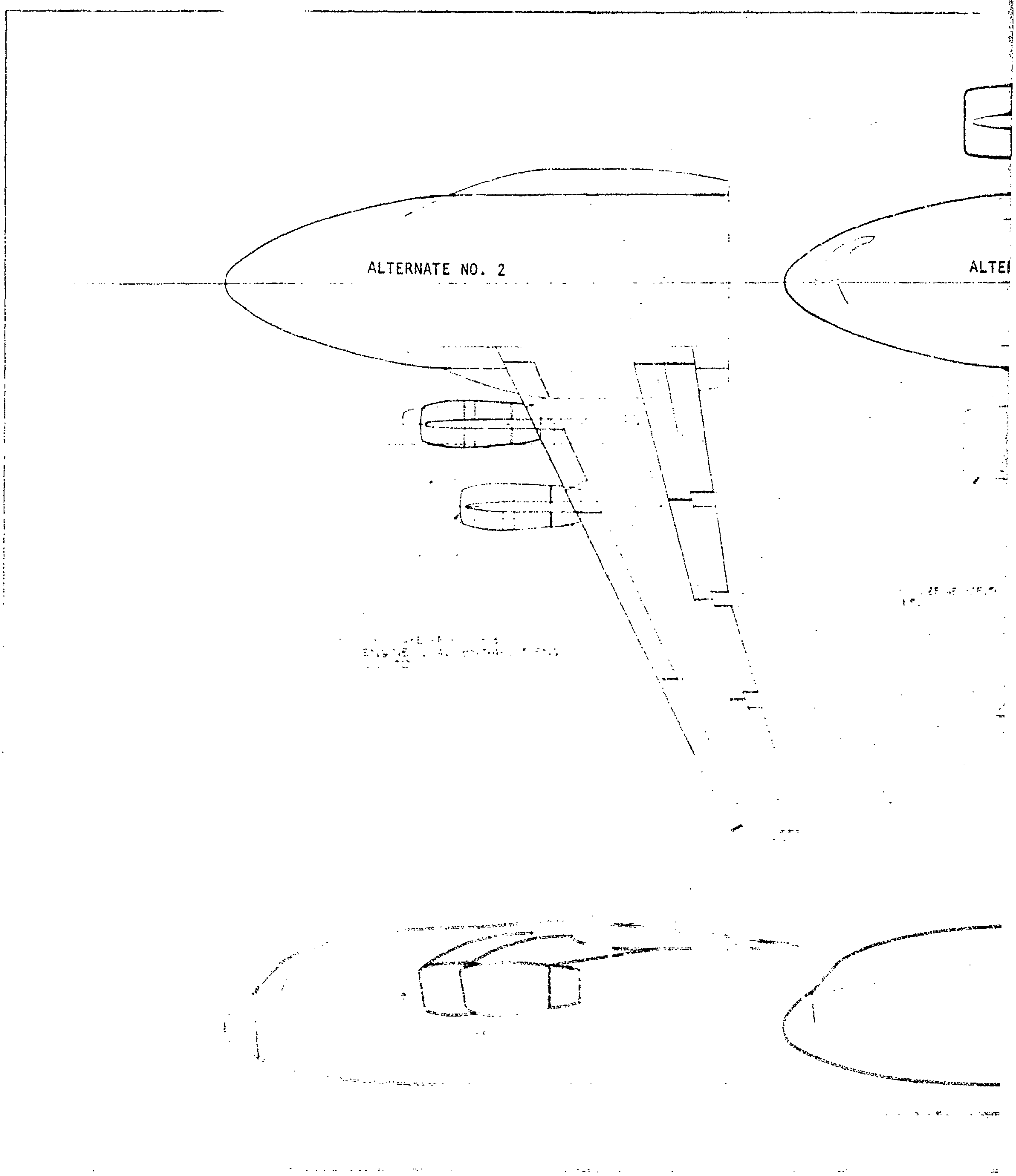
13.5.1 FLIGHT DEMONSTRATOR

The purpose of the flight demonstrator is to verify flight characteristics of a STOL transport aircraft of the anticipated size and configuration of an MST mission-oriented aircraft. The flight demonstrator program will verify, by operational experience, those vehicle flight characteristics that are required for routine short takeoff and landings. A full-size demonstrator vehicle could also serve as a test bed for an engine flight test development program and provide for practical propulsion/airframe integration of an externally blown flap tactical STOL transport. The philosophy incorporated in the STAI demonstrator discussed in this section is to use the basic vehicle developed as the refined study baseline configuration and to exchange engines, with performance being the fall out. Several candidate engines were considered, with three being selected for analysis. The flight demonstrator configuration description, propulsion, weights, performance, and program/costs/schedules are presented.

13.5.1.1 Configuration Description

The flight demonstrator conceptual design is a modified updated basepoint configuration using the basic airframe with alternate engine installations. The updated configuration descriptions may be found in section 3.0. Three engines were selected as candidates for the flight demonstrator propulsion system analysis. The propulsion systems analyzed incorporated the TF-34-GE-2, F101/F13, and TF33-P-7 engines. Two of the three propulsion systems, using the TF34-GE-2 and F101/F13 engines, are shown installed on the updated configuration in figure 133. The TF33-P-7 engine installation would be similar in size as that shown for the F101/F13 engine installation. This engine could be used as an externally blown flap propulsion system, but is considerably heavier than either the installed TF-34-GE-2 or F101/F13 engine.

Engine installation placement and nacelle size were accomplished by first removing the GE13/F10 engine nacelle and replacing it with one of the candidate engine systems. The replacement nacelle length and diameter were evaluated using the same engine-nacelle size relationships that were used on the GE13/F10 installation. The engine-nacelle length and diameter ratio relationships were held constant for wetted area and weight changes associated with each of the engines. Nacelle fore-aft placement on the wing is with the nozzle exit located at the wing leading edge. Spanwise



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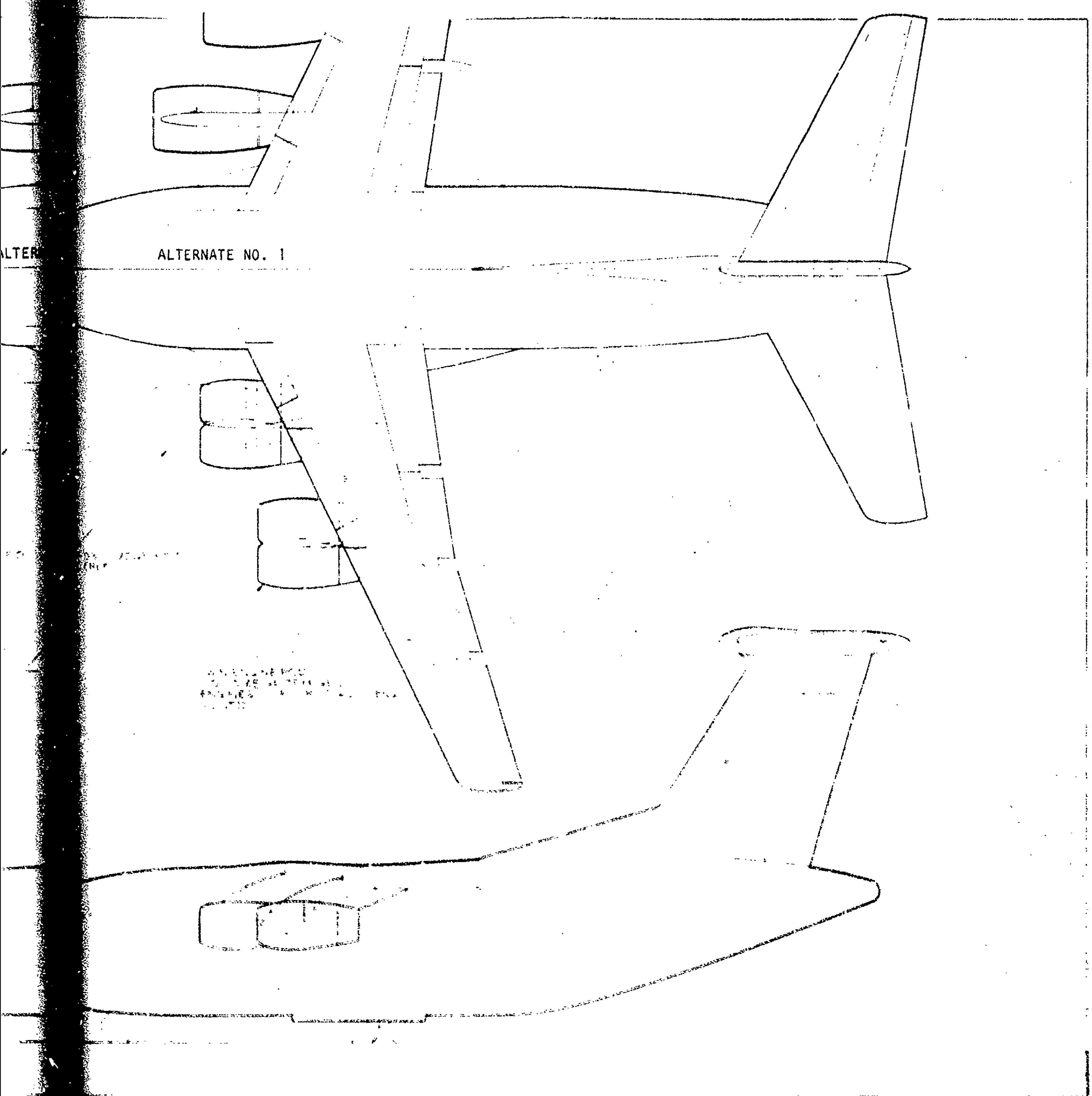

Figure 115. Flight Demonstrator Alternate Engine Installation

location is established in a similar manner as described in section 3.0. The inboard nacelle centerline is one nacelle maximum diameter from the fuselage, and the outboard nacelle is 1.5 nacelle maximum diameters from the inboard nacelle. The nacelle containing the dual-engine installation uses the maximum nacelle diameter in the vertical direction at the engine centerline for its placement from the fuselage and nacelle spacing. Dual-engine placement within a single nacelle would be similar to a B-52-type installation. Thrust reversing is accomplished by a cascade-type reverser on each side of the dual-engine nacelle. The nozzle design is similar to that used on the GE13/F10 nacelle, but would be independently operated for each of the engines in the nacelle. A summary of the engine-nacelle dimensional characteristics is given in table XLIX.

TABLE XLIX. STOL DEMONSTRATOR NACELLE DIMENSIONAL CHARACTERISTICS

	GE13/F10	TF34-GE-2	F101/F13	TF33-P-7
Diameter (in.)				
Engine	68.9	50	48.80	53.00
Nacelle	87.00	63 height, 115 width	61.6	66.9
Length (in.)				
Engine	103.1	80.20	148.24	142.00
Nacelle	191.8	149.20	275.73	264.10
Wetted area (sq ft)	340.00	420.00	345.96	359.86

13.5.1.2 Weight Analysis

The engine installation effects on the refined baseline vehicle were computed by considering those portions of the aircraft that were directly affected by the propulsion system. Structure weights were computed for modifications in nacelle wetted areas and in nacelle pylons, due to installed engine weight changes. Propulsion group weights were revised to account for the engines under consideration. All vehicles were analyzed at the baseline takeoff gross weight of 159,310 pounds, and fuel weight was

adjusted for changes associated with a particular engine installation. Additional weight consideration may be justified for the flight demonstrator by deleting various production aircraft features such as the cargo handling system, ground mobility, and some fixed equipment items, for improved flight demonstrator performance. Table L indicates the weight effects that may be expected for the three alternate engines.

TABLE L. STOL DEMONSTRATOR WEIGHT SUMMARY

	Baseline	Alternate 1	Alternate 2	Alternate 3
Engine	GE 13/F10	TF34-GE-2	F101/F13	TF33-P-7
Engine Size (%)	93	100	100	100
No. of engines	4	8	4	4
Installed engine weight	14,444	14,624	14,484	22,736
Engine section or nacelle group	4,528	4,880	4,528	6,278
Total weight empty	95,684	96,216	95,724	105,726
Fuel weight	33,161	32,630	33,121	23,119

The above weight summary does not indicate a weight penalty due to changes in materials that would be required due to increased exhaust temperature of the F101/F13 and TF-33-P-7 engines. Preliminary estimates indicate that portions of the wing and flap which are subjected to direct jet wake impingement would be constructed with the same design concept as the remainder of the flap, but titanium would be used instead of aluminum. This material change would not have a significant weight effect, but would add to fabrication problems as well as increased costs. The vehicles that would use the F101/F13 or TF-P-7 engines would have an estimated 1-percent weight increase in the use of titanium in their basic material mix.

13.5.1.3 Propulsion System

The propulsion system for the demonstrator EBF airplane will be required to have early availability, a minimum of development or installation cost, and low risk, and to meet the demonstrator requirements of the final EBF configuration design. The vehicle thrust-to-weight ratio dictates the engine thrust rating necessary to perform these design conditions, and the engines considered must be available hardware or presently in development to be made available within the desired time period.

The candidate engines used for the investigation study configuration are shown in table XIV. The engines shown in table LI were selected from these candidate engines to be representative of the possible choices for a demonstrator vehicle.

The TF33-P7 (JT3D) turbofan is an existing hardware engine of the desirable thrust rating for a four-engine installation with low cost and excellent reliability. However, it has disadvantages in application to STOL vehicles in the areas of weight, size, fuel specifics, and exhaust characteristics.

The F101/F13 turbofan is a nonaugmented version of the F101-GE-100 afterburning turbofan being developed for the B-1 aircraft. The basic engine is retained with only removal of the afterburner section to avoid additional development cost and time. The engine would offer the best advanced state-of-the-art design with minimum risk, since the B-1 engine is scheduled to complete the PFRT within a year. The F101/F13 engine thrust rating is a little low for a four-engine version of the proposed EBF configuration; therefore, the demonstrator vehicle performance would not be favorable unless the airplane could reduce in weight or size. The relatively low bypass ratio of this turbofan would also be a slight disadvantage in cruise fuel consumption and exhaust characteristics.

TABLE LI. SELECTED STOL DEMONSTRATOR ENGINES
(SEA LEVEL, STANDARD DAY RATING)

Mfg	Model	$F_{N_{MAX}}$	T/W	BPR	FPR	OPR
P&WA	TF33-P7	21,000	4.52	1.25	1.9	16.0
GE	F101/F13	16,150	5.50	2.0	2.26	26.5
GE	TF34-GE-2	9,280	6.53	6.23	1.5	20.5

The TF34-GE-2 high-bypass turbofan is presently in initial production for the Lockheed S-3A, the Fairchild Republic A-10A prototype, and the Boeing AWACS airplanes. The high-bypass engine represents the cycle characteristics that are desirable for EBF STOL vehicles, and closely matches the GE13/F10 cycle selected for the refined baseline configuration. The thrust rating is slightly low for the proposed configuration using an eight-engine installation, but an uprated version, that appears likely within a few years, should offer good vehicle performance. The exhaust characteristics make the engine attractive for an EBF airplane, from a low noise and temperature consideration. Since the TF34 is being used for sound suppression investigation and proposed prototype vehicles for STOL configurations, it would offer many installation advantages obtained from these applications or studies.

A two-engine installation was considered, using engines such as the CF6 or JT9D turbofans. Because a two-engine arrangement does not allow use of the proposed EBF configuration without a major vehicle redesign and an engine-out conditions becomes a problem, this concept appears very unattractive. Therefore, the larger size turbofan engines were not included in the candidate engine list.

Preliminary performance data were generated for the selected engines to use in the STOL demonstrator airplane performance evaluation. Installation characteristics based on the proposed configuration requirements were established to account for performance losses due to inlet total pressure recovery, inlet cowl (spillage) drag, horsepower extraction, and compressor bleed airflow. The change in external nacelle and pylon drags due to the selected engine installation and different exhaust characteristics are accounted for in the vehicle external aerodynamics performance.

Each of the selected engines is installed with similar installation characteristics. The engines have simple rounded-lip, fixed-geometry, open-nosed inlets sized to provide good performance during takeoff and to allow minimum spillage during cruise conditions. A mixed-flow engine is preferred for the EBF configuration; therefore, all engine installations used a common exhaust nozzle for the fan and gas generator exhaust. The mixed-flow exhaust system includes a thrust deflector that directs the exhaust stream into the wing flaps during lift-off or landing approach, and below the wing flaps for ground roll. The engines are installed with a cascade-type fan thrust reversing system located in the upper forward portion of the nacelle, just aft of the fan section.

13.5.1.4 Performance

The performance characteristics of the flight demonstrator with the selected alternate propulsion systems are compared to the refined baseline configuration performance capability. Design mission performance for all vehicles was evaluated at a constant takeoff gross weight and payload weight over the same mission profile. Short takeoff and landing calculations were computed using the methods presented in volume III, "Performance Methods and Takeoff and Landing Rules."

The pertinent performance characteristics for the baseline and the alternate propulsion systems are shown in table LII. All vehicles have the same initial wing loading and takeoff gross weight.

TABLE LII. STOL DEMONSTRATOR PERFORMANCE SUMMARY

	Baseline	Alternate 1	Alternate 2	Alternate 3
Engine	GE 13/F10	TF34-GE-2	F101/F13	TF33-P-7
Engine size (%)	93	100	100	100
No. of engines	4	8	4	4
Fuel wt (lb)	33,161	32,630	33,121	23,119
Initial T/W	0.516	0.434	0.382	0.497
Design mission radius (n mi) with 28,000 lb payload	500	441	377	165
Cruise mach No.	0.75	0.695	0.75	0.75
Max mach No. at 20,000 ft	0.75	0.71	0.75	0.75
Midmission T/W	0.5453	0.428	0.377	0.4527
Midmission W/S	90.7	90.6	90.4	93.4
Takeoff distance (ft)	1,240	1,180	2,680	1,790
Landing distance (ft)	1,675	1,720	2,270	2,060

Figures 134 and 135 present short takeoff and landing performance versus weight for the four vehicles. The baseline and alternate 1 configuration have similar STOL performance for the same weight range. The alternate 1 configuration thrust-to-weight ratio, with all engines operating, is lower than the baseline vehicle, but has improved performance for the one-engine-out case, since one-eighth of the thrust is lost instead of one-fourth. The eight-engine configuration lift losses and trim penalties in the STOL configuration would be less than experienced by a comparable four-engine configuration. Additional advantages of the eight-engine configuration are lower engine-out yawing moments and higher reverse thrust levels during the deceleration portion of the aborted takeoff.

13.5.1.5 Flight Demonstrator Program Schedule

The program schedule shown in figure 136 depicts the major activities of the flight demonstrator program from go-ahead through flight test completion. This schedule is for a two-vehicle flight test program that will verify flight characteristics of a STOL transport aircraft. The program will include the design, manufacturing, and flight testing of the two aircraft.

Basic engineering release is scheduled for completion at the end of the 21st month, the first aircraft completion by the end of the 31st month, and the second aircraft rollout and aircraft No. 1 first flight by the end of the 34th month. The flight demonstrator flight test program completion date would be at the end of the 52nd month after go-ahead.

The program, as shown, allows for the initiation of long-lead purchase orders as soon as practical. Some review period, before purchase order release, would be required to identify those items that would be critical to the overall program schedule. Those items critical to structural testing phase would receive priority during the early part of the review period.

The structural testing program would be initiated by conducting small element testing. These tests will be conducted early during the first design stages so that the results can benefit the initial design effort. Concurrently, small panel specimens will be designed and released for testing. Fabrication and testing of medium structural specimen for the fuselage, wing, empennage, and other critical vehicle components are phased for completion before basic release, in order for test data results to be incorporated in the design drawings. Major test sections for the wing boxes and fuselage shell structural segments will be used to evaluate the load capabilities of these airframe components. These tests are scheduled to be completed in the early stages of airframe fabrication such that any changes in design, due to test results, can be incorporated during that stage.

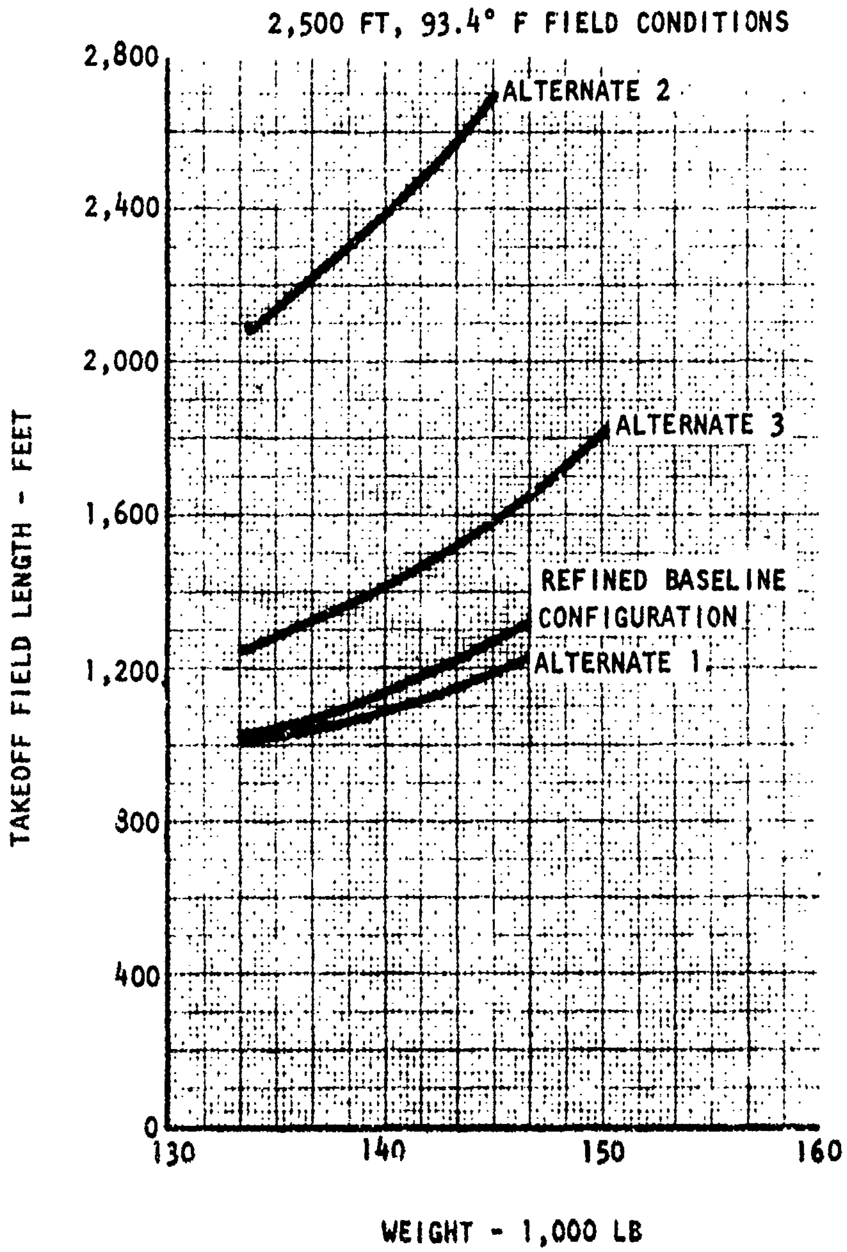

Figure 134. STOL Demonstrator Takeoff Field Length Comparison

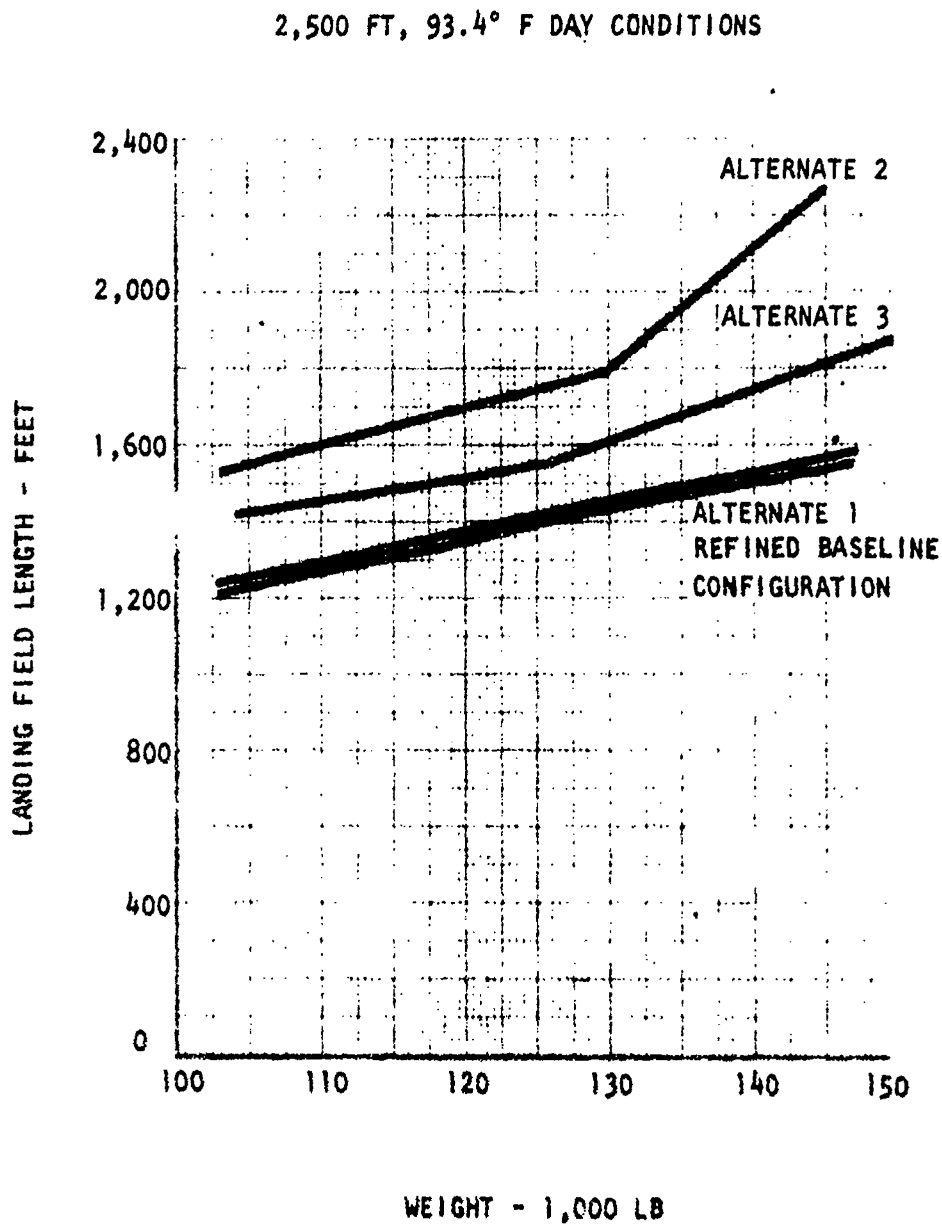

Figure 135. STOL Demonstrator Landing Field Length Comparison

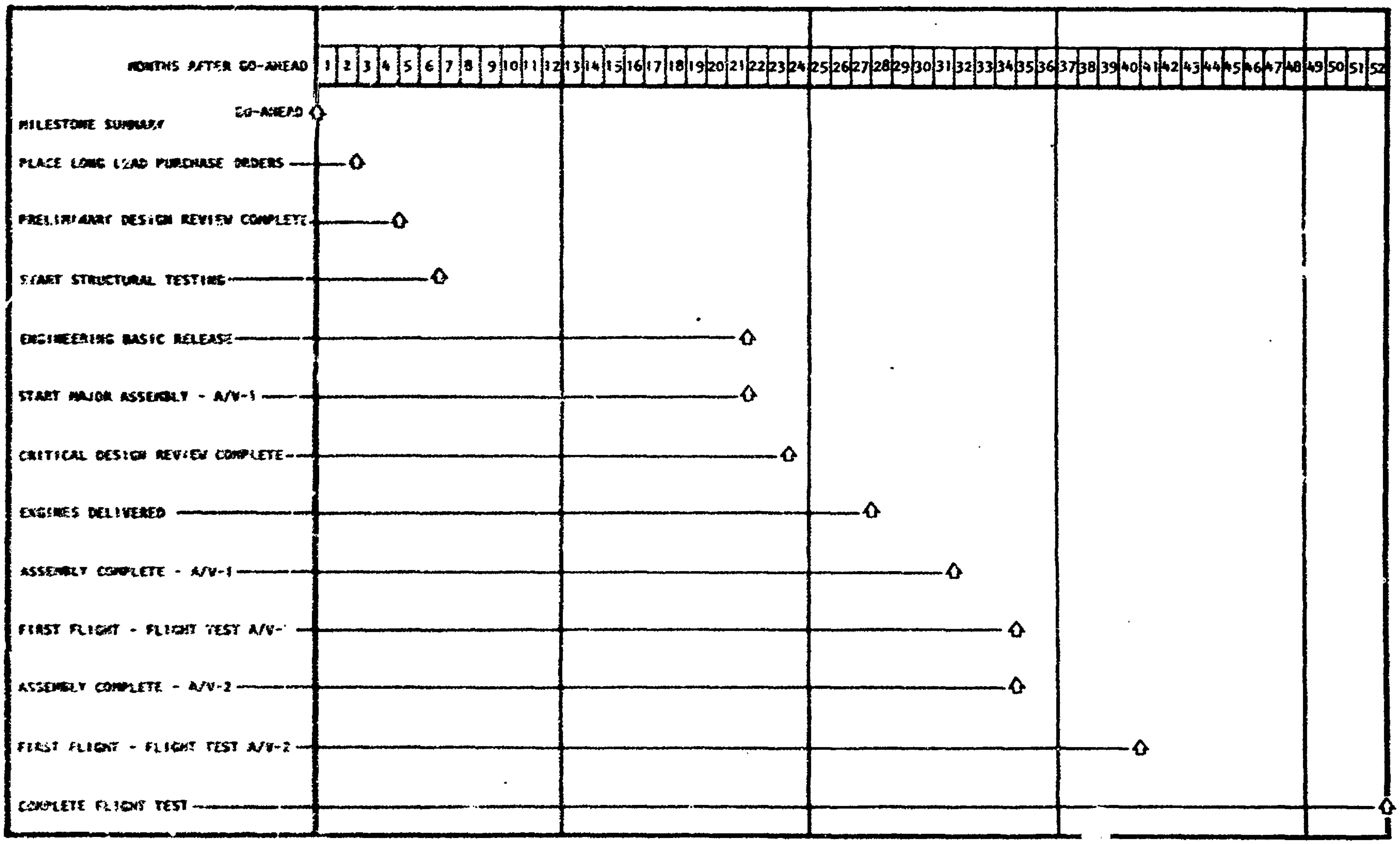

Figure 136. Flight Demonstrator Program Schedule

Engineering basic release is expected to be completed by the 21st month, along with the initiation of major assembly of aircraft No. 1. Two months later a critical design review will be completed to insure design integrity. The next major milestone is the arrival of the required engines before complete assembly of the No. 1 aircraft. Aircraft rollout is scheduled at the end of the 31st month, with the first flight scheduled starting the 35th month.

The flight test program provides for 315 hours of flight testing and is scheduled to last 18 months. The basic flight test program elements will consist of initial airworthiness tests, flight characteristics determination, performance and propulsion tests, avionics and subsystem tests, environmental test, flight loads investigations, vibrations and flutter tests, and operation flight and subsystem characteristics evaluations. Specific test tasks are flight characteristics, including stability, controllability, and handling; air vehicle performance, including STOL takeoff and landing performance; vibration and flutter; environmental control; and subsystems.

13.6 BUDGETARY COSTS AND SCHEDULES

13.6.1 ENGINEERING STUDIES

A schedule for items discussed in paragraphs 13.1 through 13.4 is presented in figure 137. Also shown are level of effort and budgetary costs.

13.6.2 FLIGHT DEMONSTRATOR

13.6.2.1 Cost Methodology

13.6.2.1.1 Material Equipment

Dollar-per-pound data from previous studies and programs were analyzed and updated to provide the basis for estimating the STOL aircraft. Material and equipment costs were estimated in constant 1974 dollar values.

13.6.2.1.2 Engineering

The man-hours required for engineering were developed by comparison of requirements with prior programs and with estimating relationships derived from these programs. Adjustments were made to include hours for effort peculiar to STOL aircraft that are not found in the historical data base.

Item	Level of Effort 1,000 Man-Hours				Budgetary Costs - $1,000			
Years	1	2	3	4	1	2	3	4
Configuration and Subsystem Design								
Configuration trades	7	7			140	140		
Alternate BLC air source	4				75			
Ground mobility system	15	16			300	325		
Nozzle deflector development	4				75			
Aerodynamic Design Techniques								
Aerodynamic control development	3				165			
Low-speed wind tunnel tests (80 hr)								
Analysis								
Performance improvements	7.2	1			420	20		
Low-speed wind tunnel tests (100 hr)								
High-speed wind tunnel tests (240 hr)								
Analysis								
Methods development	3.2				195			
Low-speed wind tunnel tests (100 hr)								
Analysis								
Structures and Mass Properties								
Criteria survey	2.5				50			
Design & test of powered lift system components	50	75	75	15	950	1400	1400	250
Application of advanced composites	10	2.5			200	50		
Weight estimation methods	4.2				80			
Flight Control Techniques								
Flight simulator tests & analyses	12	4			225	75		
Handling qualities requirements								
Piloting procedures								
Airfield requirements								
Cockpit controls & displays								
Advanced control system concepts	4	2			83	42		

Figure 137. Engineering Studies Budgetary Cost and Schedules

Supporting activities, including shop support of engineering, planning, quality control and electronic data processing, and other related items, were estimated by historical relationships to engineering labor hours.

13.6.2.1.3 Manufacturing

The manufacturing hours per pound at T_1 were estimated based upon an analysis of the structural concept design, historical data, and similar program estimates. Ratios applied to the total estimated manufacturing hours were used to estimate the man-hours required for manufacturing planning and manufacturing quality control.

13.6.2.1.4 Tooling

The tooling hours-per-pound AMPR developed from previous programs were used as a base in determining the basic tooling man-hours requirements. The impact of the structural concepts and material selection were considered in adjusting these data. Additional factors for material were then developed to constitute the total tooling requirements for this program. Ratios were applied to the total tooling hours to estimate the man-hours required for tool planning and quality control.

13.6.2.1.5 Other Rates Factors

Labor and overhead rates are based on the B-1 Division's rate projections in constant 1974 dollar values.

Procurement expense and general and administrative (G&A) expense rates used also reflect constant 1974 rates. The procurement expense rate is applied to all procured materials and services; the G&A rate is applied to all costs.

Estimated earnings are computed on total estimated costs, excluding GFAE. A 10.0-percent rate was used in computing estimated earnings for the subject program.

13.6.2.2 Pricing Considerations

1. Propulsion and avionics were considered GFAE

2. All tooling is prototype soft tooling

The estimates in table LIII are submitted for budgetary and planning purposes only.

TABLE LIII. BUDGETARY AND PLANNING ESTIMATES

Function	Alt No. 1 Flt Demo (2) (in millions)	Alt No. 2 Flt Demo (2) (in millions)
Engineering	$137.3	$147.1
Tooling	78.9	83.7
Fabrication	50.2	52.1
Total (2) flt demo	$266.4	$282.9

SECTION XIV

CONCLUSIONS

The design requirements of this study of an externally blown flap tactical transport aircraft are met with an airplane having a design takeoff gross weight of 159,310 pounds, including a 28,000-pound design payload. This vehicle meets the 500-nautical-mile design mission radius and has a 2,600-nautical-mile range deployment mission carrying a 29,031-pound payload.

The EBF tactical transport concept provides a practical means of obtaining the desired STOL performance for a MST with relatively low risk. Improvements in performance and further reduction of program risk could be obtained with additional wind tunnel testing, flight controls optimization, configuration refinements, and further development of flight control techniques and piloting procedures. Additional effort should be concentrated in the structural design aspects involving the effects of the engine exhaust impingement on the wing and flap structure.

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